FlightSafety international

CITATION EXCEL
PILOT TRAINING MANUAL

VOLUME 2
AIRCRAFT SYSTEMS

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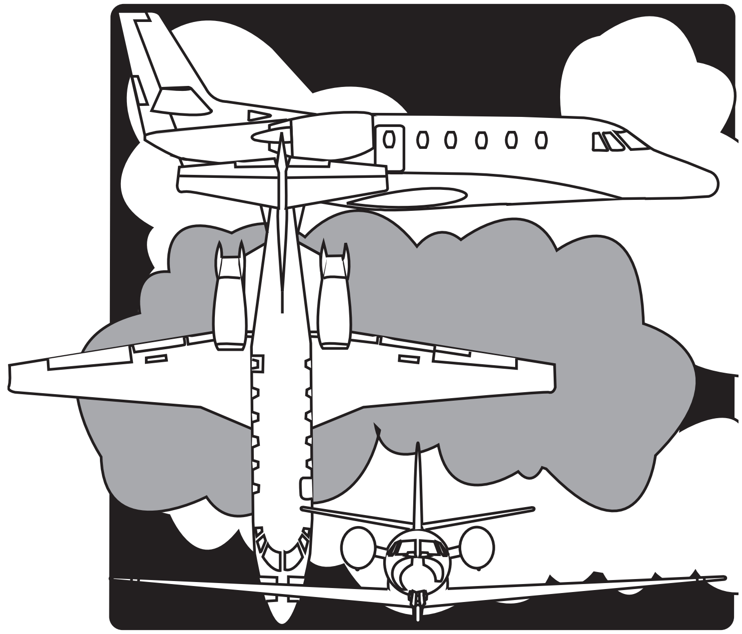
CHAPTER 1
AIRCRAFT GENERAL



INTRODUCTION

This training manual provides a description of the major airframe and engine systems installed in the Cessna Citation Excel. The information contained herein is intended only as an instructional aid. This material does not supersede, nor is it meant to substitute for, any of the manufacturer's maintenance or operating manuals. The material presented has been prepared from current design data.

Chapter 1 covers the structural makeup of the airplane and gives an overview of the systems. An annunciator section in this manual displays all annunciator and other light indications and should be pulled out for reference while reading this manual. Review questions are contained at the end of most chapters. The questions are included as a self-study aid, and the answers can be found in the appendix.

GENERAL

The Citation 560XL (Excel), UNs 5001 and subsequent, is certified in accordance with FAR Part 25 airworthiness standards and utilizes fail-safe construction concepts. It combines systems simplicity with ease of access to reduce maintenance requirements. Low takeoff and landing speeds permit operation at smaller airports. High bypass turbofan engines contribute to overall operating efficiency and performance.

The minimum crew requirements for operations in the Citation Excel are one pilot and one copilot. The pilot-in-command must have a Citation 560XL type rating and meet the requirements of FAR 61.58 for two-pilot operation. The copilot shall possess a multiengine rating and meet the requirements of FAR 61.55.

STRUCTURES

The Citation Excel (Figures 1-1 and 1-2) is a pressurized low-wing monoplane. Two Pratt and Whitney Aircraft of Canada Limited PW545A turbofan engines are pylon mounted on the rear fuselage. Figures 1-1,1-2 and 1-3 display

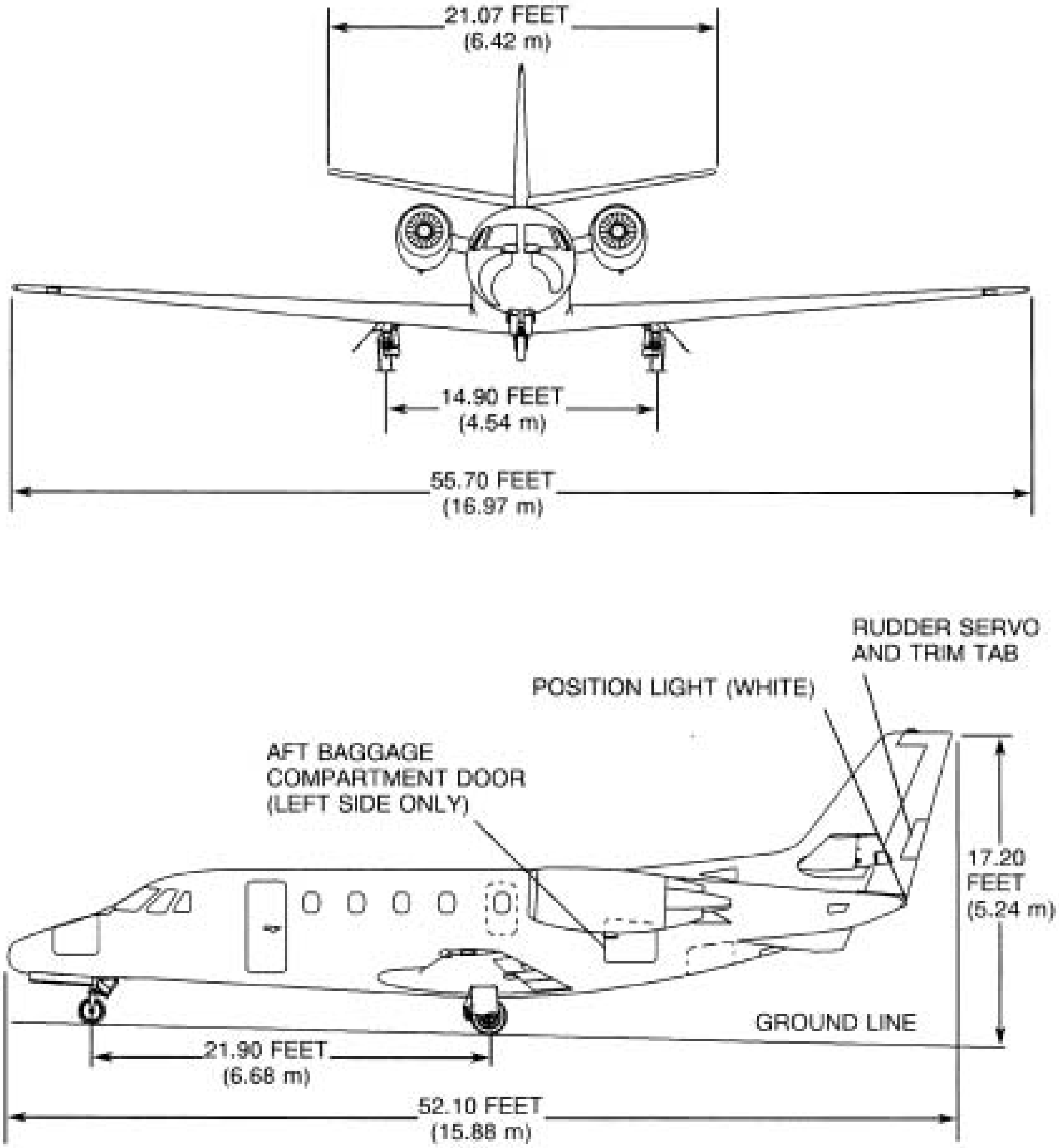


Figure 1-1 Principle Dimensions

drawings of the Citation Excel containing the approximate exterior dimensions, cabin dimensions, and turning radius.

NOSE SECTION

The nose section is an unpressurized area containing the avionics compartment, and equipment areas. Doors provide access to avionics components, brake reservoir, pneumatic bottle, etc. Each door incorporates two paddle latches, a cam key lock, and one safety "pin" latch. Each door paddle latch has an integral micro switch which will signal an annunciator panel light in the cockpit to illuminate the ACC DOOR UNLOCKED-NOSE message if any paddle latch is not secured properly on either door.

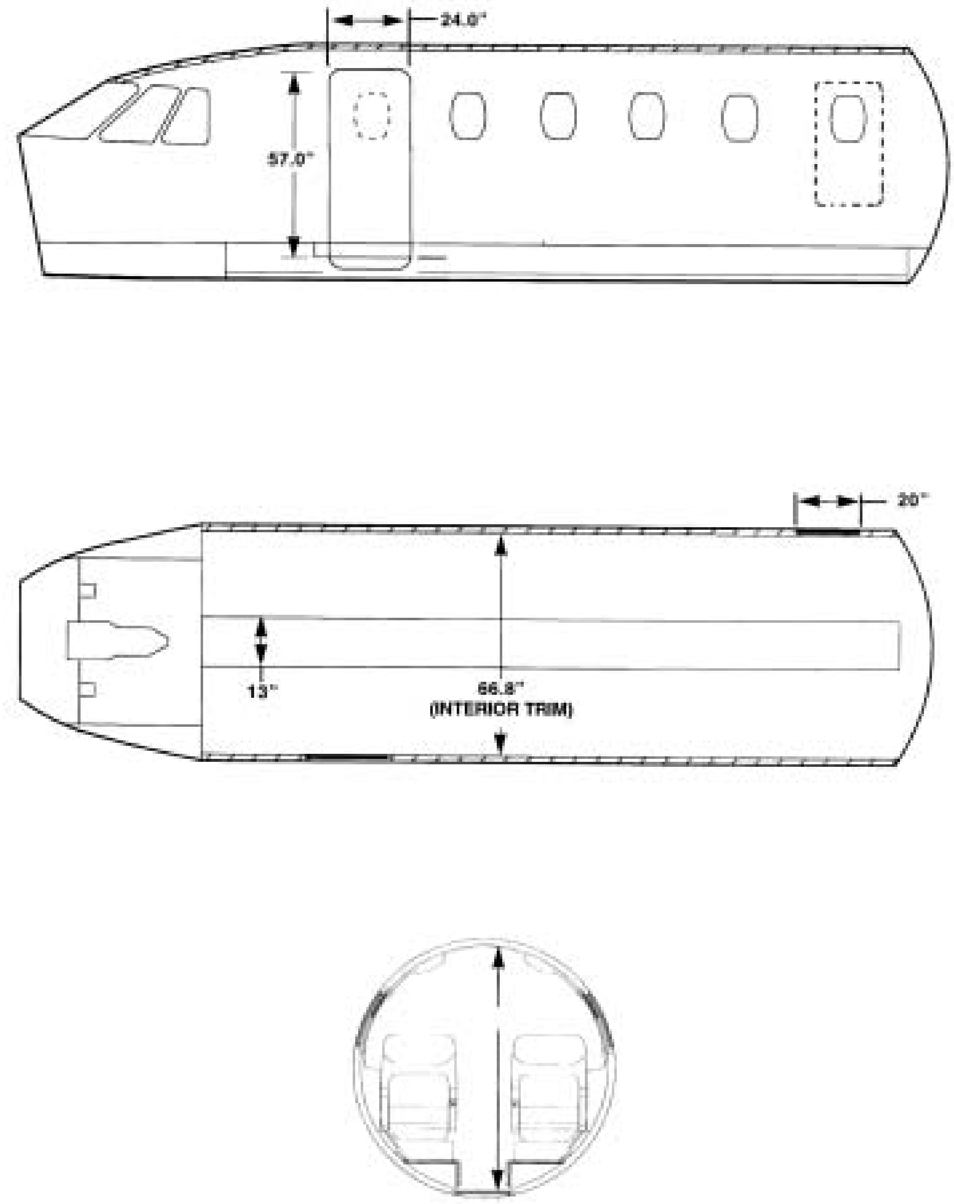
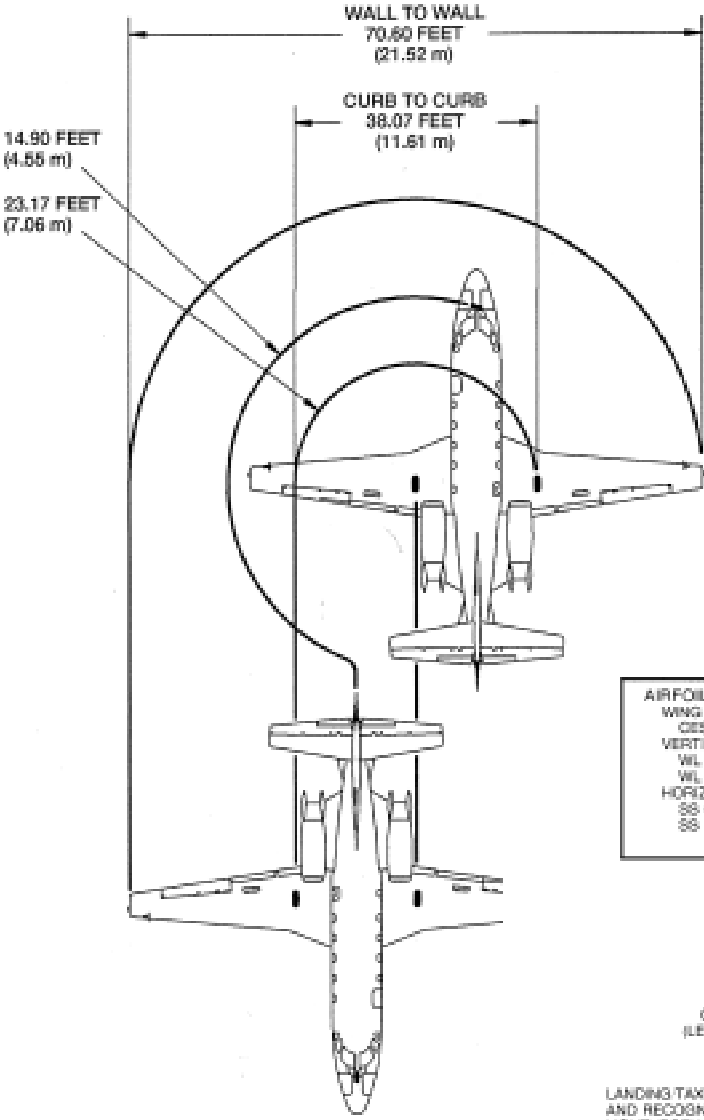


Figure 1-2 Excel Cabin Dimensions



EXCEL TURNING RADIUS AND GENERAL INFORMATION

AIRFOILS		INCIDENCE	
WING		WING	
CESSNA MODIFIED		WS 34.00	+ 3° 33'
VERTICAL TAIL		WS 335.023	-1.22'
WL 138.93	NACA 0012	2 POSITION HORIZONTAL TAIL	
WL 254.75	NACA 0008	NOSE UP	1°
HORIZONTAL TAIL		NOSE DOWN	2°
SS 0.00	NACA 0010	DIHEDRAL	
SS 126.42	NACA 0008	WING	4°
		HORIZONTAL TAIL	9°

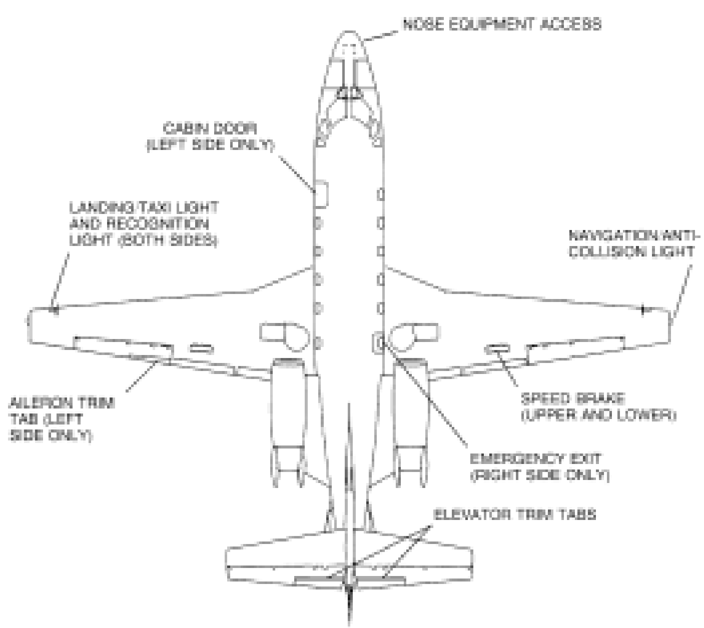


Figure 1-3 Taxi Turning Limits

CAUTION

ENSURE THE KEY IS REMOVED FROM THE CAM LOCK PRIOR TO FLIGHT TO PREVENT POSSIBLE INGESTION OF THE KEY INTO THE ENGINE.

The doors are secured directly to the nose structure by aluminum hinges and swing up to open and allow access to the nose compartments. The front hinges contain grounding straps for lightning protection. The aft hinge on each door connects to a spring loaded pneumatic cylinder that holds the doors open and over center geometry of the cylinder also holds the doors closed when unlocked (Figure 1-4).

Figure 1-4 Nose Door Pneumatic Lift Cylinder

The brake reservoir, power brake accumulator, emergency pneumatic bottle, and the digital anti-skid brake computers are located in the left nose compartment (Figure 1-5). The oxygen bottle is located in the right nose compartment. Nose mounted avionics components may be accessed through both sides with the nose compartment doors open.

CABIN

DESCRIPTION

The cabin extends from the forward bulkhead to the aft pressure bulkhead and includes the flight station, passenger seating compartment and small stowage compartments. The cabin measures approximately 24 feet in length bulkhead to bulkhead, 5 feet 7 inches in width, and 5 feet 8 inches in height. Cabin baggage compartments are located in the forward and aft passenger compartments and allow storage of such items as briefcases, coats and small suitcases. Passenger seats may have sliding drawers for additional stowage.

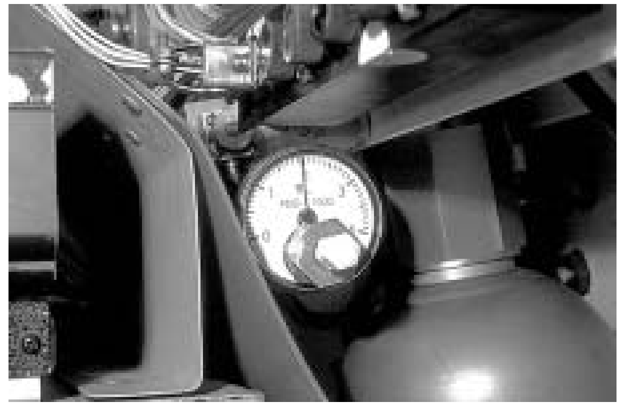


Figure 1-5 Pneumatic Bottle and Sight Gage

A typical interior seating arrangement consists of seven passenger seats (options for a maximum of 12), two flight crew seats and a rear mounted toilet seat. The passenger cabin area is equipped with overhead Passenger Service Units (PSU) which contain individual passenger controlled air outlets and reading lamps. The passenger compartment is also equipped with overhead indirect lighting, dropped aisle (footwell) lights, and lights that illuminate the entrance door area, refer to, Chapter 3, LIGHTING.

FLIGHT COMPARTMENT

Two complete flight crew stations are provided with dual controls that include control columns, adjustable rudder pedals, and wheel brakes. There are two fully adjustable seats each equipped with seat belts and two inertia reel shoulder harnesses.

There are two aft openable side windows on each side of the cockpit that open inward (Figure 1-6).

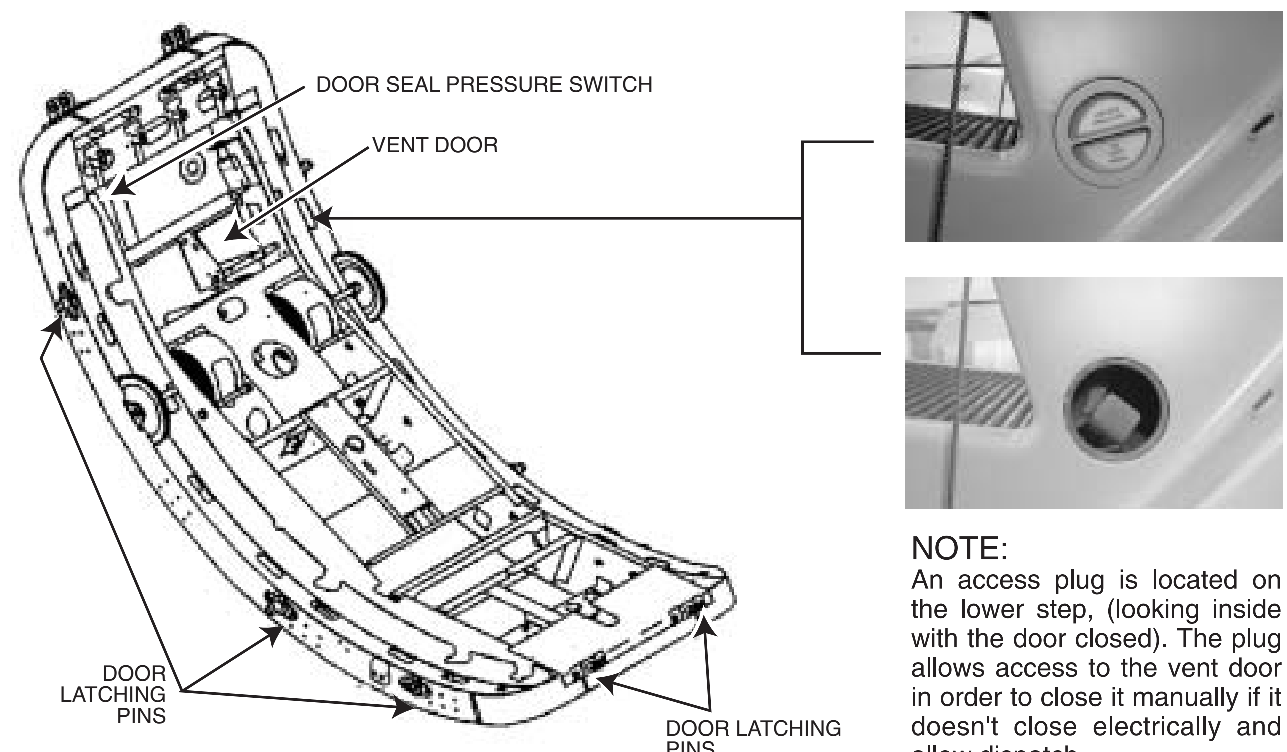
Figure 1-6 Aft "Openable" Side Windows

CABIN ENTRY DOOR

A cabin entry door located on the left side of the fuselage forward of the wing, is a one-piece air stair door hinged at the bottom that opens down and outward. The door contains solid steps on the interior side. The door is counterbalanced by a cable and spring torque bar when lowered and raised, for ease of opening and closing. The door is locked open by over center geometry. The door is secured in the closed position by eight (8) latch pins, three (3) on each side and two (2) on top, that extend from the door perimeter into latch pin plates in the door frame.

A small vent door, integral to the main door (behind the lower step), opens as the cabin door is unlocked to vent any residual cabin pressure prior to releasing the door latch pins (Figure 1-7 and 1-8). Closing the cabin door positions an external spade door "closed" to cover the lower portion of the main cabin door and mesh with the fuselage fairing to reduce drag (Figure 1-9).

Installed along the outer perimeter of the cabin door are two seals. The primary seal inflates with engine bleed air as the door is closed to induce a tight seal for cabin pressurization. If the primary seal loses pressure, a secondary pressure seal (non-inflatable) should hold cabin pressure. Bleed air inflatable acoustic seals are installed in the door frame to reduce noise.



NOTE:
An access plug is located on the lower step, (looking inside with the door closed). The plug allows access to the vent door in order to close it manually if it doesn't close electrically and allow dispatch.

Figure 1-7 Entrance Door Components

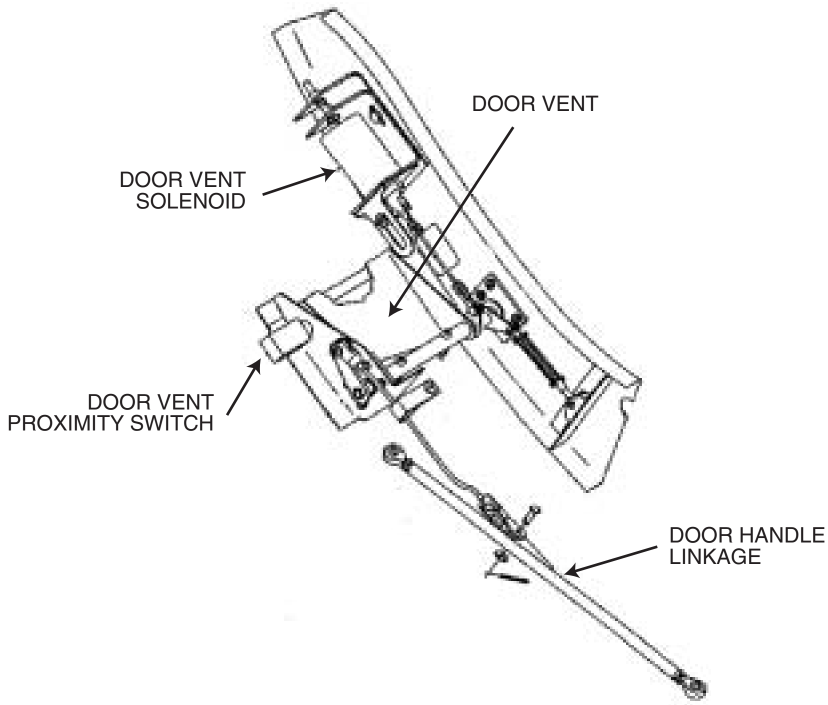


Figure 1-8 Vent Door

Figure 1-9 External Spade Door

OPERATION

The outside door handle is flush mounted and can be key locked for cabin security. The handle may be pulled outward by the finger hole in the small end and rotated 90° clockwise to unlatch the door. Rotating the door handle down should release the pre-catch assembly.

The pre-catch assembly is designed to align the door latch pins with the door frame receptacles automatically while closing the door. The pre-catch is normally released when the door handle is rotated to unlock the door. If the pre-catch does not release (door does not open slightly), depress the small round "pre-catch release" button on the fuselage directly forward of the door.

The door is pulled outward and down. At its lowest point, a foot may be placed on the lower step to push it down further. Pressing down on the handrail will lock the door over center.

Pulling UP on the handrail or pulling UP on the Raising/Lowering Handle (forward side of door) inside the cabin, unlocks the over center geometry and allows the door to be raised. From inside the cabin, the door is pulled tightly closed and the inside door handle is released from the stowed position by depressing the button on the handle to spring load it out of the stowed detent. Pulling the handle out and rotating clockwise

DOWN will extend the latching pins into the door frame and lock the door. This closing motion also closes the integral vent door and locks the exterior spade door. As the lower aft latching pin extends into the door frame receptacle, it opens the door seal inflation valves and inflates the primary door seal and the acoustic seals provided either or both engines are operating, or and APU running. The handle is stowed by rotating the handle clockwise back to the stowed position.

Opening the door from the inside, the door handle is released from its stowed position and rotated counter-clockwise DOWN to pull the latching pins into the door perimeter. The door should open slightly. If the door doesn't open, pulling the pre-catch release lever (red handle) adjacent to the front side of the door (directly opposite the exterior release button), will release the pre-catch. Pushing the lever back in will restow the pre-catch release.

Unlocking the cabin door will open the vent door, unlock the spade door and open the door seal deflation valves allowing the door seals to deflate. Pushing out on the door will start the door down, and pushing DOWN on the Raising/Lowering handle will continue the door down until it locks over center. The handrail is connected to the raising/lowering handle and extends as the door is lowered. The over center locking linkage, the two telescoping support struts and handrail, provide solid support for entering and exiting the cabin via the cabin door steps.

MONITORING

Inspection windows in the interior side of the cabin door near the door handle and each latch pin allow for visual inspection of the locking mechanisms. A green flag in the windows indicate proper pin engagement and proper locking of the handle.

The **CABIN DOOR** annunciator panel light is connected to electrical switches that monitor the two upper latch pins and two lower latch pins on the forward and aft sides of the door, the door handle lever locking pin, and the vent door is closed electrically by a solenoid and monitored by a proximity switch. The following sequence must be satisfied to extinguish the **CABIN DOOR** annunciator.

1. Two upper and two lower door latch pins fully extended into the door frame.

2. Door handle locking pin is in the locked position.

3. Vent door is closed - electrically closed by a solenoid and pull cable (Figure 1-8).

NOTE

If the **CABIN DOOR** annunciator remains illuminated after the door is secured, a small indicator panel with six lights located directly below the cabin entry light switch on the forward cabin door frame (Figure 1-10) will indicate one or more "red" lights illuminated (unlocked). The six indicators will normally extinguish if all microswitches associated with the four door latch pins are fully extended into the door frame, the door handle is locked properly and the vent door is closed. The indicator lights located top left and right, and bottom left and right monitor four door latch pins. The top center and bottom center indicators monitor the position of the door handle and vent door respectively.

Figure 1-10 Cabin Door Indicators

NOTE

If the cabin door is closed and locked prior to placing the BATT switch ON, and the **CABIN DOOR** annunciator illuminates, place the BATT switch OFF for a few seconds an then ON to extinguish the light (or unlock and relock the door). Locking the door with the BATT switch ON activates a 0.5 second timer to electrically close the vent door after all other micro switches are satisfied.

If the BATT switch is placed ON and then OFF, the timer is active for 30 minutes (battery bus power). Therefore, if the door is closed and locked within this 30-minute period with the BATT switch OFF, placing the switch ON will not illuminate the **CABIN DOOR** light.

The **DOOR SEAL** annunciator light will illuminate if the door seal is not inflated (less than 5 psi). The annunciator will extinguish when door seal pressure exceeds approximately 8 psi. Locking the door extends the lower aft latching pin into the door frame and opens the door seal pressure valve which allows service air pressure (Chapter 9, PNEUMATIC SYSTEM) to inflate the seal.

NOTE

If the primary door seal deflates in flight, **DOOR SEAL** annunciator illuminates flashing, the secondary bayonet type pressure seal should prevent the cabin from depressurizing.

The acoustic seals around the door frame are not monitored by annunciator warning.

EMERGENCY EXIT

An emergency, over-wing escape hatch is installed into the right side of the fuselage (Figure 1-11). It is a plug-type door installation and has a provision for inserting a locking pin to prevent unauthorized entry while the airplane is on the ground. The pilot must ensure that the pin is removed prior to flight.

Figure 1-11 Emergency Exit Door

The emergency exit door can be opened from either outside or inside the airplane. If the emergency exit door is not properly secured, the **EMER EXIT** annunciator will illuminate "flashing".

To open the escape hatch from inside the cabin:

1. Remove the inside handle cover.
2. Pull down the door release D-Handle.
3. Rotate the D-Handle clockwise until the latch pin has fully released.
4. Pull on the D-Handle to rotate the door inward about its bottom edge until the two plug pins at the bottom of the hatch disengage.
5. Place the hatch in a location that will not interfere with passenger and crew egress.

To open the hatch from the outside:

1. Push the outer door release handle to unstow the handle.

2. Rotate the handle counter-clockwise until the latch pin has fully released.

3. Push on the handle to rotate the door inward about the bottom edge until the two plug pins at the bottom of the hatch disengage and the door is removed.

4. Place the hatch in a location that will not interfere with passenger and crew egress.

Turning either the inner or outer release handle retracts the latch pin away from the proximity switch and sends a signal to the **EMER EXIT** annunciator panel light.

WING

Fuel storage is incorporated in the wings. Integral left and right fuel tanks are separated by a center rib. The fuel tank cavity extends from the center of the aircraft outboard towards the wing tip and is bounded by the forward and aft wing spars except where it is interrupted by the wheel well (Figure 1-12). Each wing is a fuel

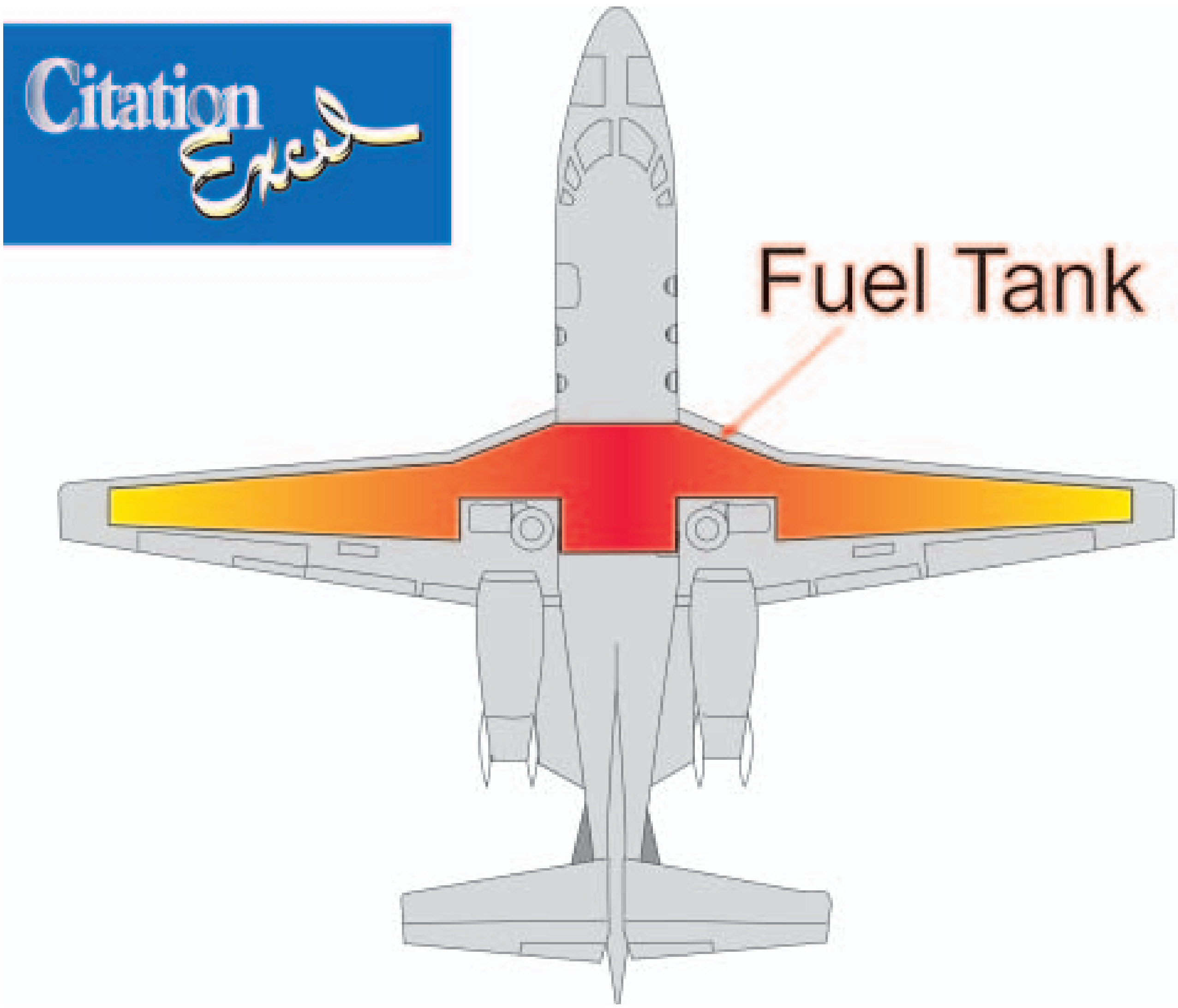


Figure 1-12 Fuel Tank Locations

tank. A Single Point Pressure Refuel/Defuel (SPPR) system access door is located on the right side of the fuselage directly in front of the right wing root (Figure 1-13).

Figure 1-13 Single Point Pressure Refuel/ Defuel Door (SPPR)

NOTE

The SPPR door is **not** connected to a warning annunciator light.

The leading edge of the wings are anti-iced by bleed-air heat (Figure 1-14). There are eleven (11) boundary layer energizers (BLEs) and a stall strip attached to the leading edge of each wing. There are twenty six (26) vortex generators attached to the top side of each wing (Figure 1-14). Speedbrakes and flaps are attached to each wing (Figure 1-15).

Figure 1-15 Wing Trailing Edge

EMPENNAGE ACCESS DOORS

The tail cone compartment is an unpressurized area that contains major components of the hydraulic, pneumatic, environmental, electrical, engine fire extinguishing system, and the optional APU, if installed. It also contains an unpressurized baggage compartment.

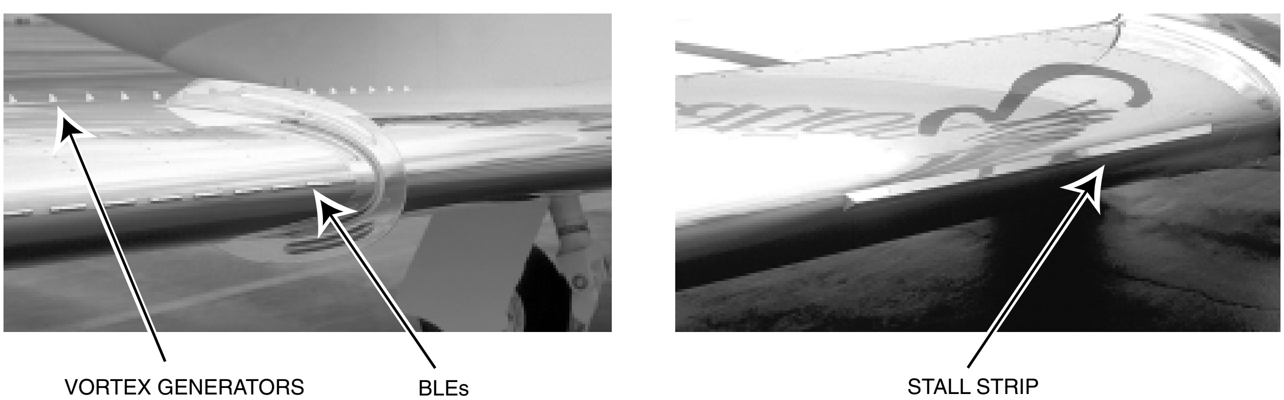


Figure 1-14 Wing Leading Edge

TAIL CONE ACCESS DOOR

The Tail Cone Access Door is located on the right side of the fuselage below the right engine nacelle (Figure 1-16). The door is equipped with a key operated cam lock for additional security and prevents unauthorized entry. Five (5) independent covered door pin “push type” latches are installed around the perimeter of the door. The door is hinged at the front and swings open when all latches are released. An electrical plunger switch mounted along the lower aft edge of the door is contacted by locking the lower aft door latch, and extinguishes the **ACC DOOR UNLOCKED-TAIL** annunciator and a light inside the tail cone area if inadvertently left on.

Figure 1-16 Tail Cone Access Door

BAGGAGE COMPARTMENT DOOR

The Baggage Compartment Door located below the left engine pylon is secured with four independent operating pin latches and a key operated cam lock (Figure 1-17). One upper pin latch assembly is equipped with a micro switch to illuminate the **ACC DOOR UNLOCKED-TAIL** annunciator if the latch is not secured properly. The door is equipped with spring loaded cylinders that allow the door to be lowered slowly if baggage is against the door (Figure 1-17). The door is equipped with steps for ease of entry. The baggage compartment can accommodate a max weight capacity of 700 lbs.

Figure 1-17 Tail Cone Baggage Compartment

BATTERY ACCESS DOOR

The Battery Access Door located forward and below the baggage compartment rotates down to open (Figure 1-18), and allows access to the main aircraft battery. The door is secured by four latches and a key-operated cam lock.

Figure 1-18 Battery Access Door

NOTE

S/N 5188 and on; the battery access door is monitored by the **ACC DOOR UNLOCK-TAIL** annunciator.

HYDRAULIC SERVICE DOOR

The Hydraulic Service Door, located forward of the tail cone access door and below the aft right wing root, is hinged at the bottom and secured with two trigger latches installed near the top of the door. This door allows access to hydraulic connection ports to attach a hydraulic service unit (Figure 1-19).

NOTE

The Hydraulic Service door **is not** connected to an annunciator warning light.

An extended hydraulic drain mast is located below the hydraulic service door to vent any excess hydraulic fluid overboard and prevent the fluid from spraying on the fuselage (Figure 1-19).

An optional externally serviceable flush toilet access door is installed directly forward of the hydraulic service door. The door swings down to open and is secured with three paddle latches (Figure 1-19).

NOTE

The optional serviceable flush toilet access door is not connected to an annunciator warning light.

Directions for servicing the toilet are placarded on the inside of the service door.

EMPENNAGE

The empennage consists of a vertical stabilizer, a two-position horizontal stabilizer, two strakes on the lower portion of the fuselage, and a dorsal fin. (See Figure 1-20).

Figure 1-20 Empennage

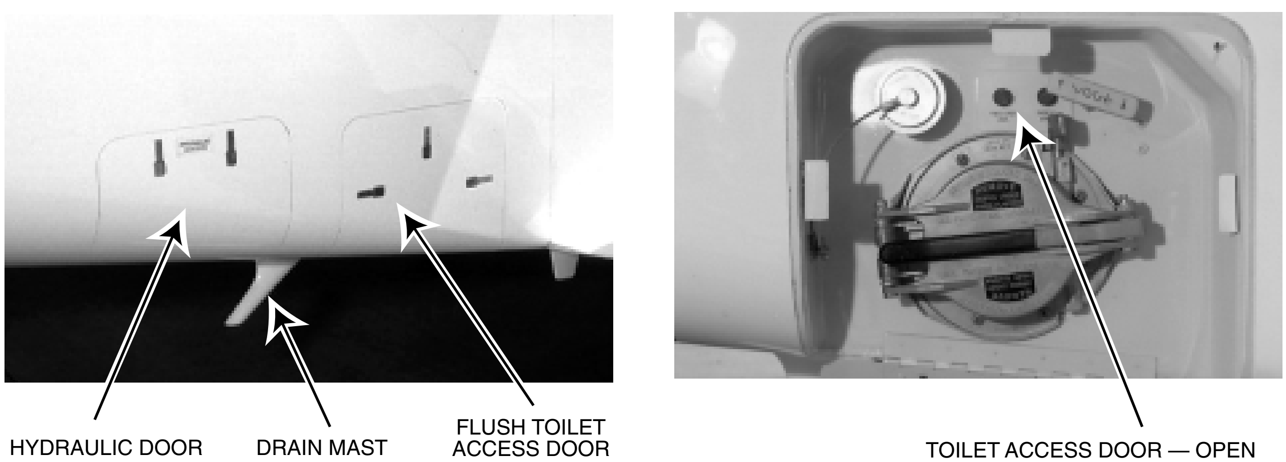


Figure 1-19 Hydraulic and Toilet Access Doors

The leading edges of the horizontal stabilizers are deiced by rubber boots. The dorsal fin, attached to the top side of the rear fuselage, has a ram-air duct to provide ram air for cooling various components in the tail cone.

ELECTRICAL SYSTEM

The airplane utilizes “DC” electrical for all normal and emergency power requirements. All DC buses are normally powered by two engine-mounted starter-generators. Engine starting power is available from either a battery or an external source. The battery is also utilized as a source of backup emergency power.

FUEL SYSTEM

The fuel system has two distinct, identical halves. Each wing tank stores a maximum of 503 U.S. gallons each (3,395 lbs each, total fuel 6,790 lbs), and supplies fuel to its respective engine. Crossfeed capability is incorporated. All controls and indicators are located in the cockpit. The aircraft is provided with a Single-Point Pressure Refueling/Defueling system (SPPR). An access panel is located on the right-hand side of the fuselage directly in front of the right wing. The SPPR door is hinged on the front side and secured with two latches and a key operated cam lock (Figure 1-21).

NOTE

The SPPR door **is not** connected to a warning annunciator.

ENGINES

Two Pratt and Whitney PW545A turbofans, installed on pylons mounted on the rear fuselage, produce 3,804 pounds of thrust each. Ice protection, fire detection, and extinguishing systems are incorporated. Target-type thrust reversers are individually operated by conventional "piggyback" controls mounted on the throttles. Refer to Figure 1-22 for intake and exhaust warning areas.

ICE PROTECTION

Ice protection for the wings is provided by engine bleed air.

The leading edges of the horizontal stabilizers are protected by deicer boots.

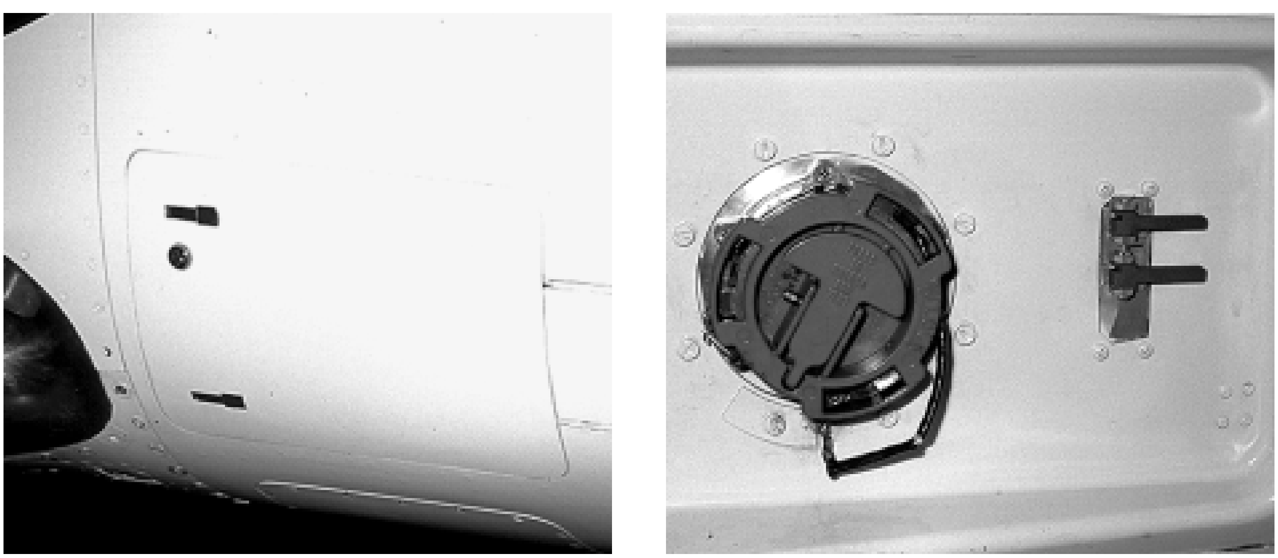


Figure 1-21 Single-Point Pressure Refueling/Defueling (SPPR) Panel

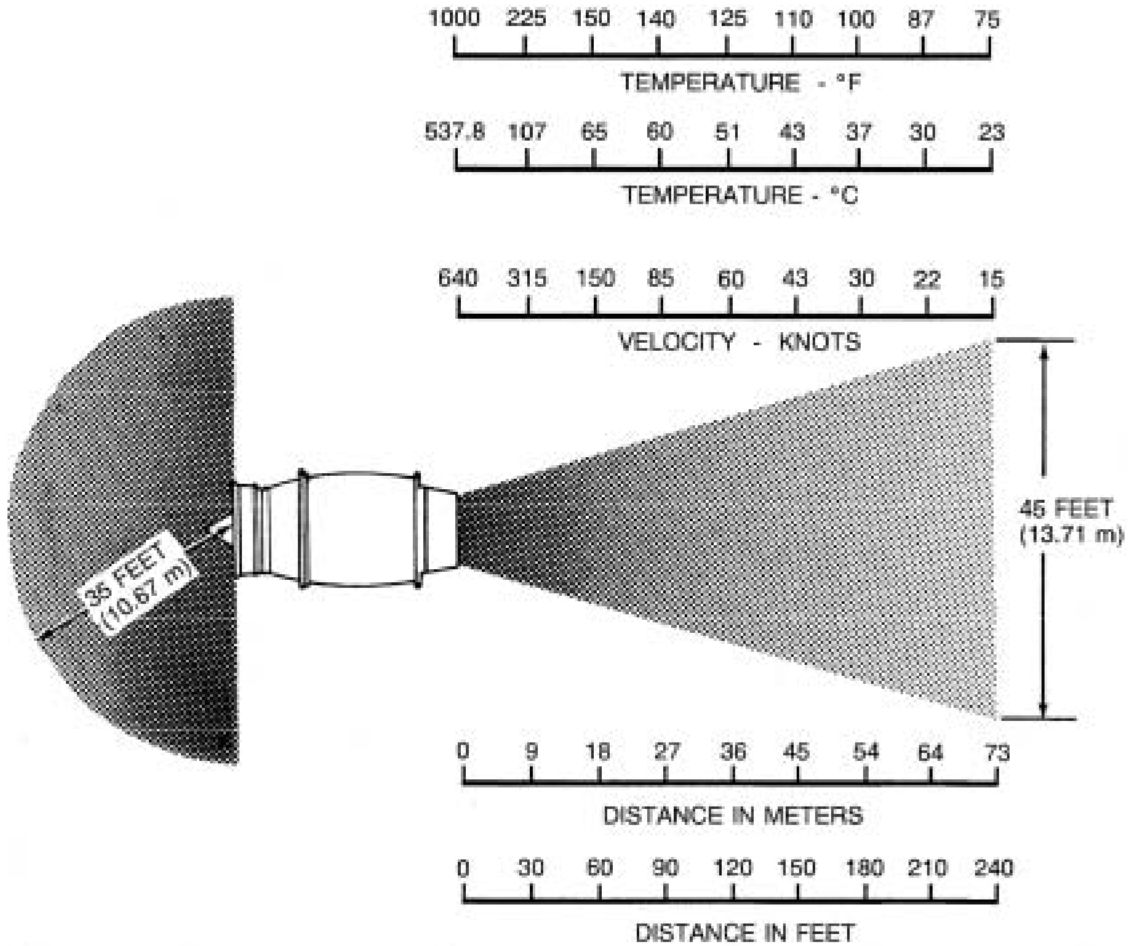


Figure 1-22 Intake and Exhaust Hazard Areas

Engine bleed-air heat from the high-speed compressor provides primary anti-ice protection.

The engine nose cone, nacelle inlet, T_1 temperature probe, and the first set of stator vanes are heated by bleed air.

The primary engine temperature (T.O.) probe is heated electrically.

Electrical heaters are employed by the pitot-static system and the angle-of-attack probe.

Engine-driven alternator power is used to electrically heat the windshield. The windshield anti-ice system is controlled by normal DC power.

All bleed-air and electrical anti-ice systems should be turned on prior to operation in visible moisture when the indicated ram air temperature (RAT) is +10°C or below.

HYDRAULIC SYSTEM

Engine-driven pumps supply hydraulic pressure for operation of the landing gear, speedbrakes, flaps, thrust reversers, and the two-position horizontal stabilizer through an open center system. The landing gear are equipped with antiskid-controlled wheel brakes, operated hydraulically from a separate enclosed hydraulic system.

Pneumatic backup is available for landing gear extension and emergency braking.

FLIGHT CONTROLS

Primary flight control is accomplished through conventional cable-operated surfaces.

Manual trimming is provided to move aileron, elevator, and rudder tabs. The elevator trim tabs are both mechanically and electrically actuated.

Speedbrakes are hydraulically operated and installed on the upper and lower wing surfaces.

Two hydraulically-operated flap segments are installed on the trailing edges of each wing.

Nose wheel steering is mechanically controlled by the rudder pedals.

The two-position horizontal stabilizer is controlled automatically with flap handle movement.

ENVIRONMENTAL CONTROL

Cabin pressurization utilizes bleed air, from the engines, which is conditioned by an Air-Cycle Machine (ACM).

Cabin and cockpit temperatures are controlled separately.

The system provides sufficient pressure to maintain sea level pressure up to an approximate altitude of 25,247 feet, and approximately 6,800-feet cabin pressure at a cruise altitude of 45,000 feet. Normal pressure is based on a pressure differential of approximately 9.3 psi.

The oxygen system supplies the cockpit through quick-donning masks stored in containers adjacent to each crew seat. And the cabin through dropout masks stored in the overhead that automatically deploy at a cabin altitude of 14,500 ft with the system in AUTO.

VAPOR CYCLE AIR CONDITIONER

A standard vapor cycle air conditioner is installed to supplement cold air from the ACM if required.

NOTE

If an optional Auxiliary Power Unit (APU) is installed, the vapor cycle unit will be removed.

AVIONICS

The standard, factory-installed avionics package includes weather radar, dual altitude reporting transponders, and a Primus 1000 integrated flight director system which incorporates the autopilot. Communication is provided by two VHF transceivers. Navigation equipment includes a digitally tuned ADF, dual DMEs, and VOR/localizer/glide-slope/marker beacon receivers. The instrument panel consists of dual EFIS and a multifunction display on the center instrument panel. ELT, CVR, and Flitefone VI are standard equipment (Figure 1-23).

A Universal, UNS-1Csp, flight management system (FMS) installed on the center pedestal is standard equipment (Figure 1-24).

A Honeywell FMZ FMS may be installed as an option.

PUBLICATIONS

The FAA *Approved Flight Manual (AFM)* is a required flight item. It contains operating limitations, operating procedures, performance data pertinent to takeoffs and landings, and weight and balance data. It does not contain enroute performance information. The *AFM* always takes precedence over any other publication.

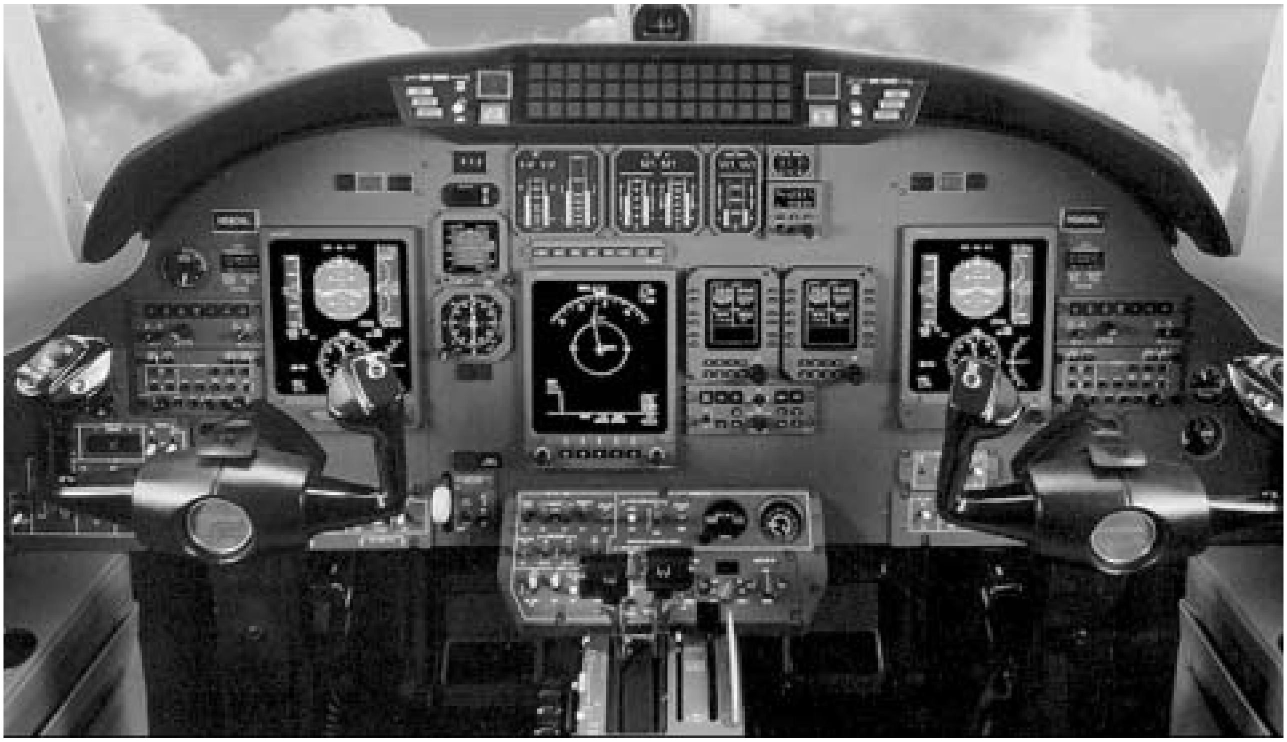


Figure 1-23 Citation Excel Cockpit

Figure 1-24 Universal UNS-1C (sp)

The *Citation 560 EXCEL Operating Manual* contains expanded descriptions of the airplane systems and operating procedures. It contains enroute flight planning information as well as climb, cruise, and descent data.

The *Cessna 560 EXCEL Abbreviated Checklist* contains abbreviated operating procedures and performance data. If any doubt exists, or the conditions are not covered by the checklist, the *AFM* must be consulted.

The *Citation 560 EXCEL Weight and Balance Manual* contains detailed information in the form of tables and diagrams. However, it is not required to be in the airplane. The basic empty weight, moments and arms are all contained in the *AFM* in order to calculate CG locations.

The *Honeywell Primus 1000 Integrated Avionics System Pilot's Manual* for the Citation Excel is a required flight item. It contains operating procedures for operating the two Primary Flight Displays, Display Controllers and the

Multifunction Display. Complete operational procedures for operating the Flight Directors and Autopilot Controller are included.

The Universal Flight Management Operator's Manual is a required flight item. It describes operational procedures for use of the Flight Management System (FMS) (Figure 1-24), its equipment, capabilities, and its operation. It details how to initialize, select and build a flight plan, and navigate using the various installed standard or optional sensors; such as VPU, GPS, LORAN C System, or INS/IRS.

The UNS-1Csp, single or dual installation, can be utilized with optional UNILINK, AFIS or GENESYS air show system. The Universal UNS-1Csp FMS is approved under TSO C129 S1/B1/C1 and meets the requirements for the following operations:

- Oceanic/Remote.
- North Atlantic (NAT) Minimum Navigational Performance Standards (MNPS).
- Enroute and Terminal RNP5/BRNAV.
- Non-Precision Approaches.

LIMITATIONS

GENERAL

The limitations presented in this chapter focus primarily on the basic airframe limits of the airplane. Specific system limitations are presented at the end of most chapters in this manual. Refer to the *AFM* for complete listings of limitations.

Certification and operational limitations are conditions of the type and airworthiness certificates and must be complied with at all times as required by law.

CERTIFICATION STATUS

The Citation 560XL (Excel) is certified in accordance with FAR 25.

AIRPLANE CONFIGURATION CODES

AA — Airplanes 5001 and on.

AB — Airplanes with rudder bias.

AC — Airplanes without rudder bias.

WEIGHT LIMITATIONS

Ramp..20,200 lbs

Takeoff ....20,000 lbs

Takeoff weight may have to be reduced to meet climb requirements or takeoff field length per *AFM*, Section IV.

Landing ....18,700 lbs

Landing weight may have to be reduced to meet climb requirements, brake energy, or for landing distance per *AFM,* Section IV.

Zero Fuel Weight (ZFW) ....15,000 lbs

Minimum Flight Weight ....12,400 lbs

Maximum Tailcone
Baggage Weight ....700 lbs

OPERATING LIMITATIONS

Types of Operation

Aerobatic maneuvers and spins are prohibited.

No intentional stalls permitted above 25,000 feet.

The airplane is approved for day and night, VFR, IFR flight and flight into known icing conditions. The airplane is not approved for ditching under FAR 25.801.

Altitude

The maximum operating altitude is 45,000 feet.

Ambient Temperature Limits

Maximum Ambient Temperature ....ISA+39°C

Minimum Ambient Temperature for Takeoff and Landing...........................-30°C

See Takeoff/Landing/Enroute Temperature Limits in Figure 1-25.

Takeoff and Landing Operational Limits

Takeoffs and landings are limited to paved runway surfaces.

Takeoff from a wet runway, when using thrust reversers for performance credit, is limited to a minimum runway width of 75 feet.

Altitude ....14,000 feet

Tailwind component................................10 knots

The autopilot and yaw damper must be OFF for takeoff and landing.

Prior to takeoff, the elevator trim check in Section III, NORMAL PROCEDURES, of the *AFM* must be satisfactorily completed.

The lavatory doors must be latched open for takeoff and landing.

Maximum Tire Ground Speed ....165 knots

Engine Sync must be OFF for takeoff and landing.

ANTISKID must be operational for takeoff.

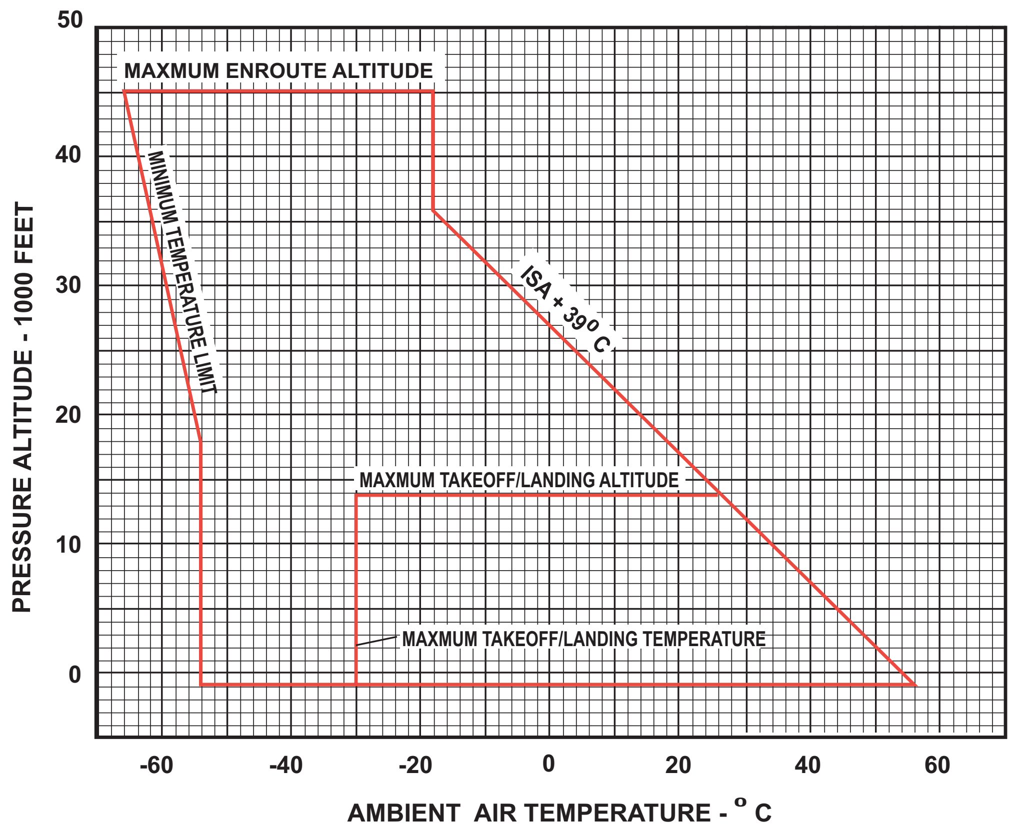


Figure 1-25 Takeoff/Landing/Enroute Temperature Limits (Ambient Air Temp is Indicated R.A.T. Adjusted for RAM Rise)

Up to three vortex generators may be missing for dispatch provided the aircraft is limited to FL410 for enroute operations. There are typically a total of 52 vortex generators installed, 26 per wing.

All Boundary Layer Energizes (BLE) must be present for dispatch (11 per wing).

Rudder bias and the rudder bias heater must be operational for takeoff, and a satisfactory preflight test must be performed in accordance with Section III, Normal Procedures in the *AFM* (AB configuration).

Crosswind Component

Without thrust reversers — Demonstrated crosswind component to 24 knots (not limiting).

With thrust reversers — Demonstrated crosswind component to 24 knots (not limiting).

Enroute

Minimum airspeed for sustained flight in icing (except approach and landing) ....160 KIAS

Maximum operating altitude ....45,000 feet

Maximum operating altitude with 1-3 vortex generators missing.....41,000 feet

Temperature ....Refer to Figure 1-25

Generator load......................................300 amps

Maneuver Limitations

No acrobatic maneuvers, including spins, are approved. No intentional stalls are permitted above 25,000 feet. Maximum maneuvering speeds are shown in Figure 1-27.

Minimum Crew

Minimum flight crew for all operations is one pilot and copilot.

Load Factors

In Flight

Flaps ....UP position 0 (-1.2 to +3.0g at 20,000 lbs)

Flaps — T/O and T/O and APP (7°) to the LAND position (7° to 35°) ....0.0 to +2.0g at 20,000 lbs

NOTE

These accelerations limit the angle-of-bank in turns and the severity of pull-up maneuvers.

Landing

Flaps — Landing ....0.0 to +2.0g at 18,700 lbs

NOTE

These accelerations limit sink rate of 600 FPM.

Weight and Balance Data

The airplane must be operated in accordance with the approved loading schedule.

Refer to the Weight and Balance Data Sheet and Model 560XL Weight and Balance Manual.

Passenger Compartment

For all takeoffs and landings, seats must be fully upright and outboard, and passenger seat belts and shoulder harnesses must be fastened. Maximum number of passenger seats is twelve (12). The lavatory door must be latched open for taxi, takeoff and landing.

Prolonged Ground Operation

Continuous engine ground static operation up to and including five minutes at takeoff thrust is limited to ambient temperatures not to exceed ISA +39°C.

Electrical load is limited to 200 amps per generator during ground operations. Transients up to 250 amp are permissible up to four minutes.

Limit ground operations of pitot-static heat to two minutes to preclude damage to the pitot-static tubes and AOA vane.

Avionics ground operation is limited to 30 minutes at ambient temperatures from 45°C to 54°C. Refer to Chapter 16, Figure 16-19.

CENTER OF GRAVITY

Refer to Center-of-Gravity Moment Envelopes in Figure 1-26:

AIRSPEED LIMITATIONS

CONDITION SPEED

V_{MO} (below 8,000 ft)..........................260 KIAS

V_{MO} (8,000 to 26,515 ft) ....305 KIAS

M_{MO} (above 26,515 ft).......................0.75 Mach

V_A ....Refer to Figure 1-27

V_{FE} 35° (Full Flaps)............................175 KIAS

Flaps extended to 7° or 15° ....200 KIAS

V_{LO} (gear retraction)...........................200 KIAS

V_{LO} (gear extending) ....250 KIAS

V_{LE} (gear extended)............................250 KIAS

Speedbrake Operating Speed V_{SB} ....No Limit

V_{MCA} (determined at maximum takeoff thrust and weight).......................90 KIAS

V_{MCG} (determined at maximum takeoff thrust and weight):

AC Configuration (without rudder bias) ....98 KIAS

AB Configuration (with rudder bias)...........................81 KIAS

Maximum tire ground speed ....165 knots

Autopilot operation........305 KIAS or 0.75 Mach

Minimum airspeed for sustained flight in icing (except for approach and landing)...160 KIAS

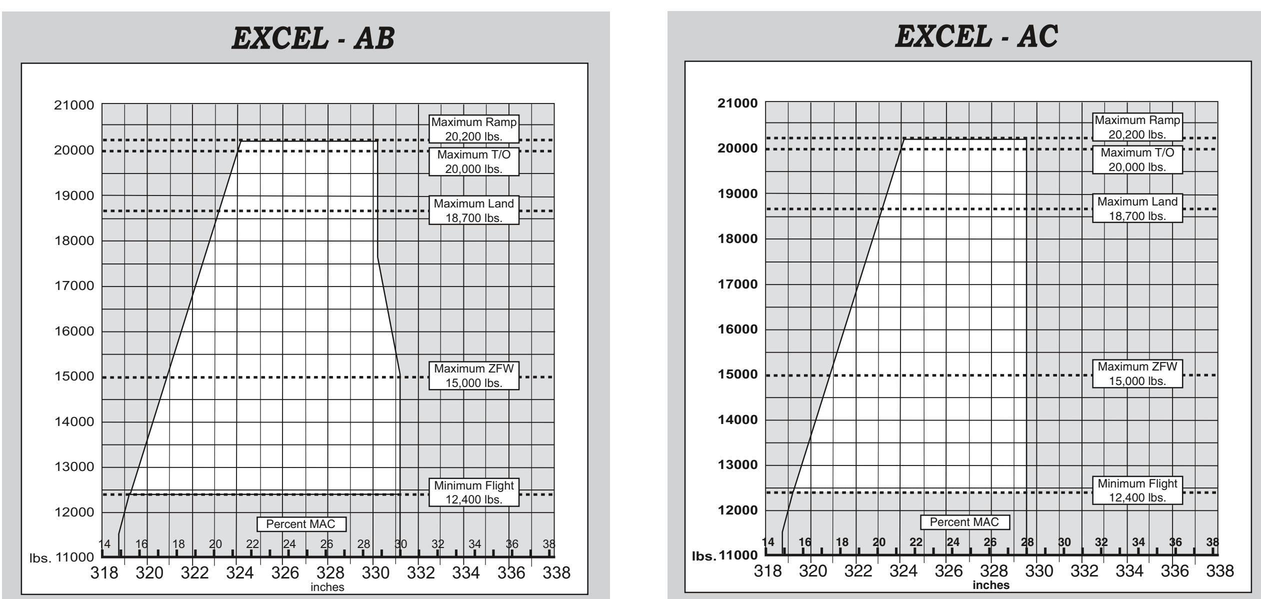


Figure 1-26 Center-of-Gravity Envelopes

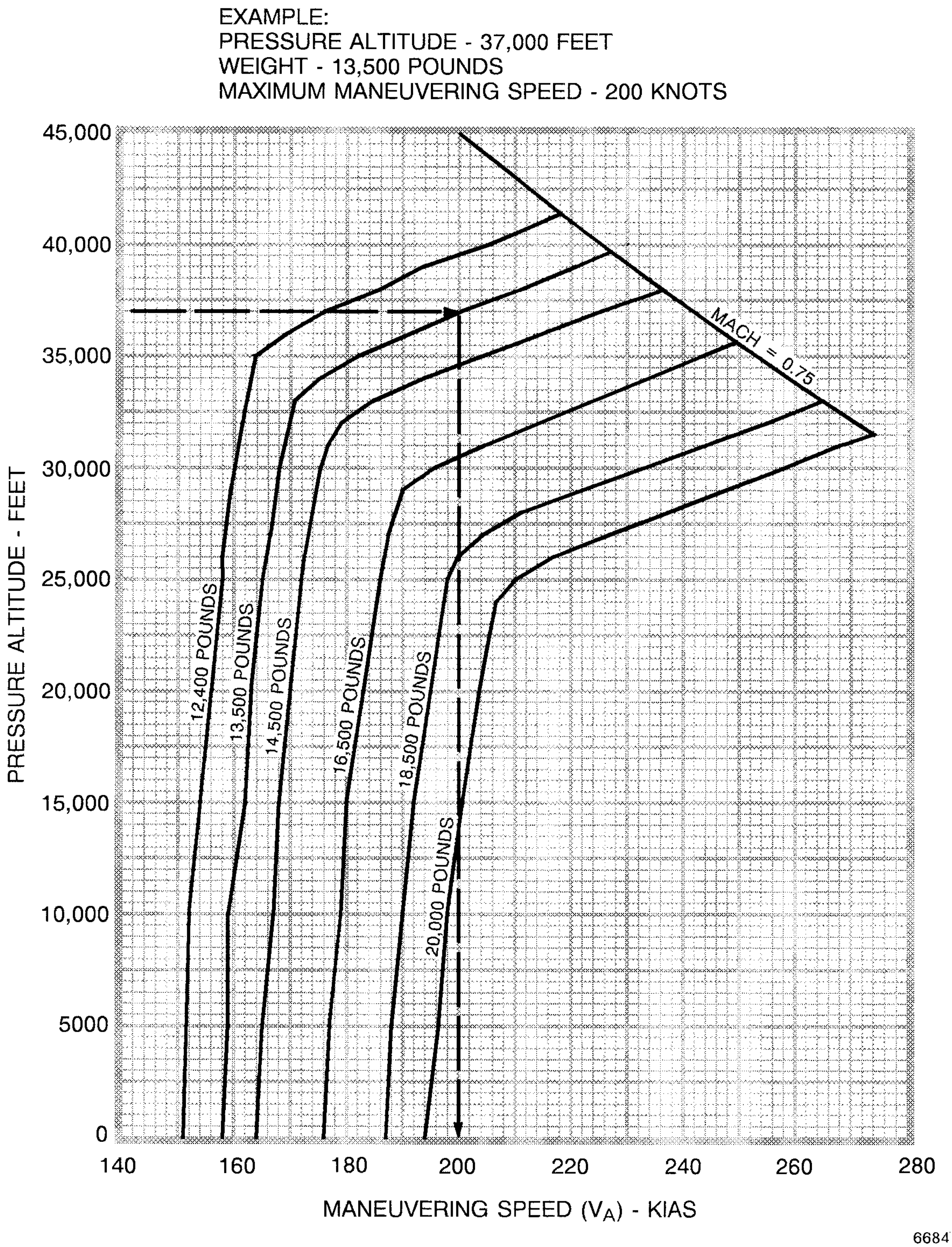


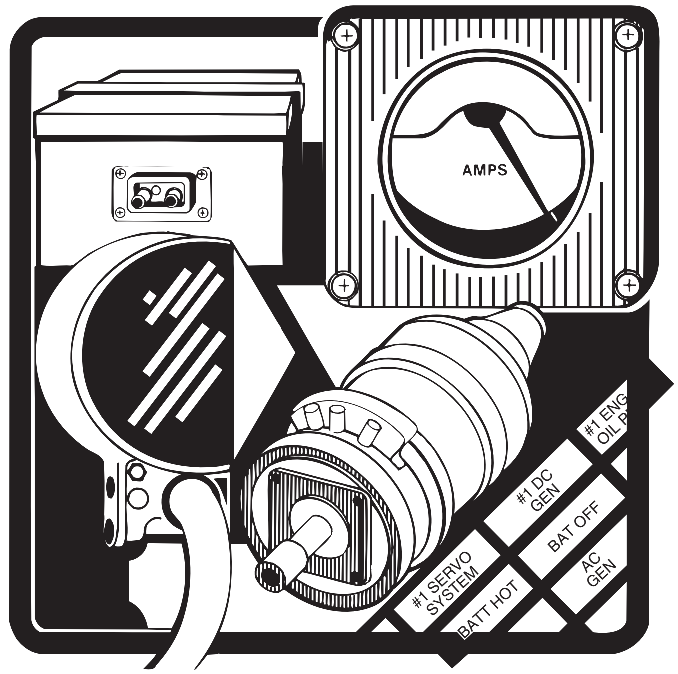
Figure 1-27 Maximum Maneuvering Speeds (V_A) — KIAS

CHAPTER 2
ELECTRICAL POWER SYSTEMS

CONTENTS

ILLUSTRATIONS

CHAPTER 2
ELECTRICAL POWER SYSTEMS



INTRODUCTION

The Citation Excel uses DC electrical power to control and/or operate various relays, valves and pumps associated with normal and abnormal systems operations. Electrical power is also an essential requirement for proper engine control and avionics displays. This chapter will describe the various sources of electrical power used by the Excel, the electrical distribution system, control switches and electrical monitoring.

GENERAL

The Excel generates its own engine-driven electrical power. It can receive and use electrical power from an optional onboard auxiliary power unit, or a tail mounted battery. Additionally, the Citation Excel can make use of an external electrical power source for ground operations.

The electrical system consists of various electrical power sources, a power distribution system, and monitoring systems.

The system is monitored by annunciator lights and gauges. Although annunciator logic is not discussed in detail until Chapter 4, the normal and abnormal annunciations that pertain to the electrical system, will be covered in this chapter.

The electrical system is predominantly direct current (DC). There are some alternating current (AC) requirements but they are very limited.

A single AC inverter powers the electroluminescent panel lighting (see Chapter 3, LIGHTING).

Two engine-driven alternators provide AC power for windshield heating applications. The alternators are discussed in detail in Chapter 10, Ice and Rain Protection.

POWER SOURCES

Power sources for the DC system include: battery, engine-driven starter-generators, an optional APU generator, and external power. Each of these power sources are discussed below. Locations for each of the power sources is shown in Figure 2-1 below.

GENERATORS

The primary source of DC electrical power is provided by two, 30-volt, 300-amp starter/ generators, mounted on the engines. The generators are controlled by Generator Control Units (GCU) located in the tailcone.

The GCUs control the generators at 28.5 volts, protect the generators and provide load-sharing capability during normal operations. When both generators are on-line simultaneously, each generator will provide approximately one-half of the total electrical load.

MAIN AIRCRAFT BATTERY

In addition to the two DC generators, a standard 44-ampere-hour, nickel-cadmium (NICAD) battery is installed with provisions for an optional Lead-Acid battery if desired. The battery is

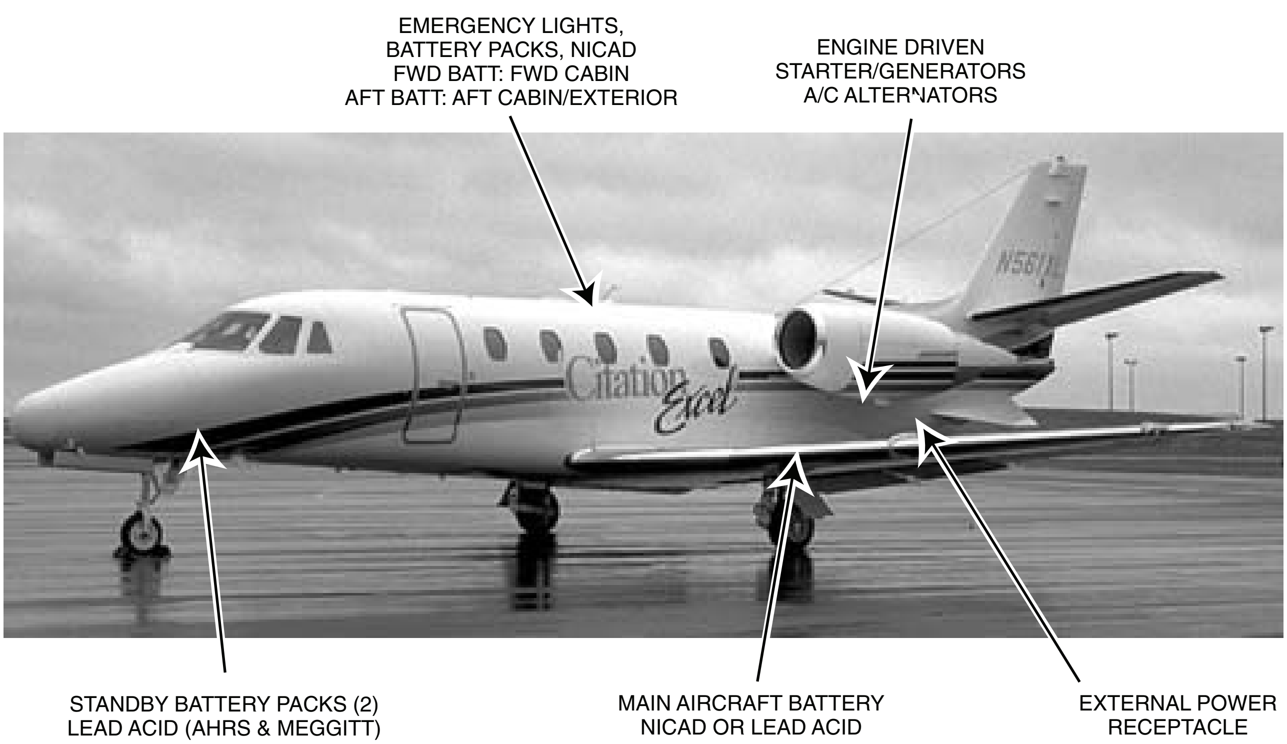


Figure 2-1 Power Source Locations

mounted inside a dedicated compartment and accessed through a door on the left side of the fuselage just behind the wing fairing (Figure 2-2). The battery is connected to the battery bus by a manual "quick connect/disconnect knob" mounted on the battery case (Figure 2-2).

Figure 2-2 Battery Compartment

Battery power only may be used for short periods, normally on the ground, for engine starting, and as an emergency power source during inflight operations.

The battery is limited in its ability to satisfy all aircraft electrical requirements. If operating on battery power only, the electrical system is designed for the crew to manually shed the majority of the electrical load to prolong battery life. This procedure becomes necessary if both generators are inoperative and the battery is the only source of DC power.

Load shedding allows the battery to provide power to critical systems for a limited time (approximately 30 minutes or more), otherwise, the battery will only power the entire electrical system for approximately 10 minutes.

Battery temperature is monitored by a temperature gauge installed in the right side instrument panel. An annunciator warning, "red" BATT O'TEMP/>160, is provided if battery temperature becomes excessive. The annunciator will flash and illuminate the MASTER WARNING flashers.

NOTE

If an optional Lead-Acid battery is installed, the annunciator warning is disconnected, but the battery temperature gage remains operational.

GROUND POWER UNIT (GPU)

Ground power can be connected to the DC distribution system through an external receptacle located on the left side of the aft fuselage (Figure 2-1). The GPU should be regulated to 28 volts and have enough amperage capability to carry whatever load is demanded, normally 1000 amps for engine starts. A GPU with soft start capability is preferred. The battery should be disconnected if GPU power is ON for a prolonged period of time.

An external power relay electrically closes to allow the GPU to connect to the battery bus. If GPU voltage is excessive or a generator is online, the relay will trip open.

NOTE

It is advisable to connect the GPU to the aircraft "hot" with voltage stabilized. This procedure protects sensitive items associated with avionics and also protects the battery if the GPU does not have reverse current protection.

NOTE

The GPU will charge the battery regardless of the position of the BATT switch.

OPTIONAL APU GENERATOR

The optional APU generator will provide 28.5 volt DC power to the entire electrical system and also parallel with the engine-driven generators. Refer to Chapter 6, AUXILIARY POWER UNIT

EMERGENCY BATTERIES

Two small size emergency lead-acid battery packs are installed in the nose compartment as a backup source of power for essential avionics. The packs are trickle charged from main DC power.

The battery pack in the right side nose compartment is dedicated to the Attitude Heading Reference Systems (AHRS).

The battery pack in the left nose compartment is the backup power source for the Standby Flight Display (SFD), or normally referred to as the Meggitt EFIS display.

There are two Nicad battery packs located in the cockpit/cabin area. They are used as a source of power for the emergency exit lights (interior and exterior). One pack is located in the cockpit and one located in the aft cabin. Refer to Chapter 3, LIGHTING, for specific information on this system.

POWER DISTRIBUTION

Electrical power is distributed from the power sources, i.e., generators, battery or GPU, to the electrical busses by various relays normally controlled by switches in the cockpit.

Refer to Figures 2-3 and 2-4: The Battery or GPU is connected directly to the BATTERY BUS and from this bus, power is supplied to the CROSSFEED BUS via the Battery Isolation Relay.

The Battery Isolation Relay is controlled by the Battery switch located on the pilot's switch panel (Figure 2-4). The relay is "closed" with the battery switch in the BATT ON position and "open" with the switch either OFF or EMER.

From the CROSSFEED BUS, electrical power flows through 225 amp current limiters to the LH and RH FEED BUSSES.

From the FEED BUSSES, power is distributed to the LH cockpit circuit breaker panel via 60 amp current limiters (3 on each feed bus), and 50 amp circuit breakers (6 on the LH CB panel).

Electrical power is sent to the RH cockpit circuit breaker panel via RH and LH Avionics (AVN) Busses, through two 60 amp current limiters on each AVN BUS and four 50 amp circuit breakers.

The Avionics busses are powered by avionics relays (AVN PWR RLY), controlled by the AVIONICS POWER ON/OFF switch (Figure 2-4).

The Emergency Bus is normally powered from the CROSSFEED BUS via an EMER PWR RLY between the CROSSFEED BUS and the EMERGENCY BUS (Figure 2-3). The relay is de-energized "closed" with the BATT switch ON or OFF.

Placing the BATT switch to EMER will energize "open" the EMER PWR RELAY between the CROSSFEED and EMER busses, and energize the EMER PWR RELAY between the BATTERY and EMER busses "closed".

Emergency Busses - Electrical power from the EMER BUS is distributed to the LH and RH CB panels in the cockpit.

Emergency power to the RH CB panel from the EMER BUS is distributed via the EMER AVN BUS through an AVN EMER RELAY and a 25 amp CB. The AVN EMER RELAY is de-energized "closed" with the AVIONICS POWER switch, ON or a loss of main DC power.

Emergency power to the LH CB panel from the EMER BUS is distributed via a 25 amp CB.

NOTE

The EMER BUS system is designed to ensure there is always a source of electrical power for critical flight instruments and various controls to safely fly the airplane to a successful landing.

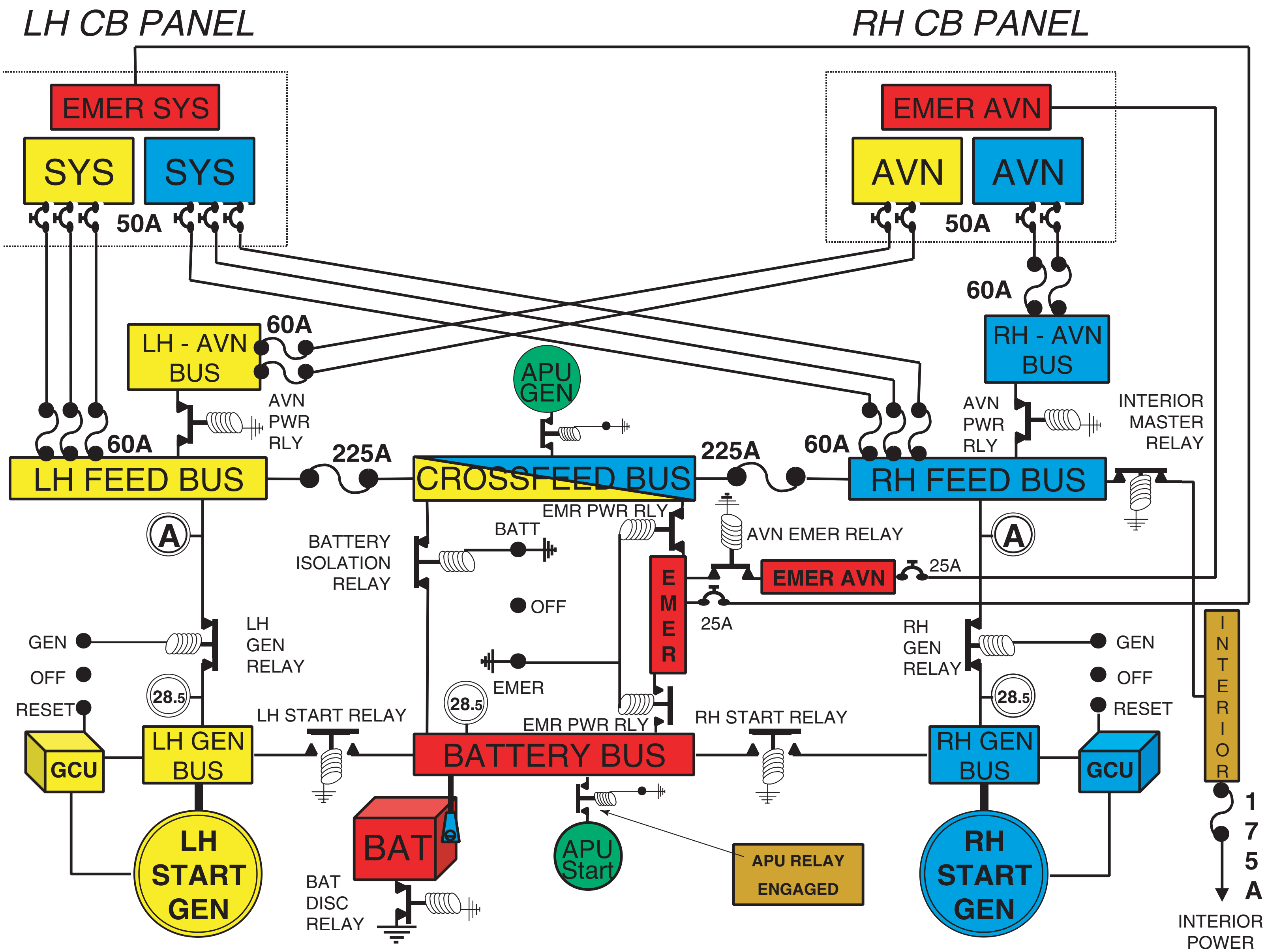


Figure 2-3 Electrical System Diagram

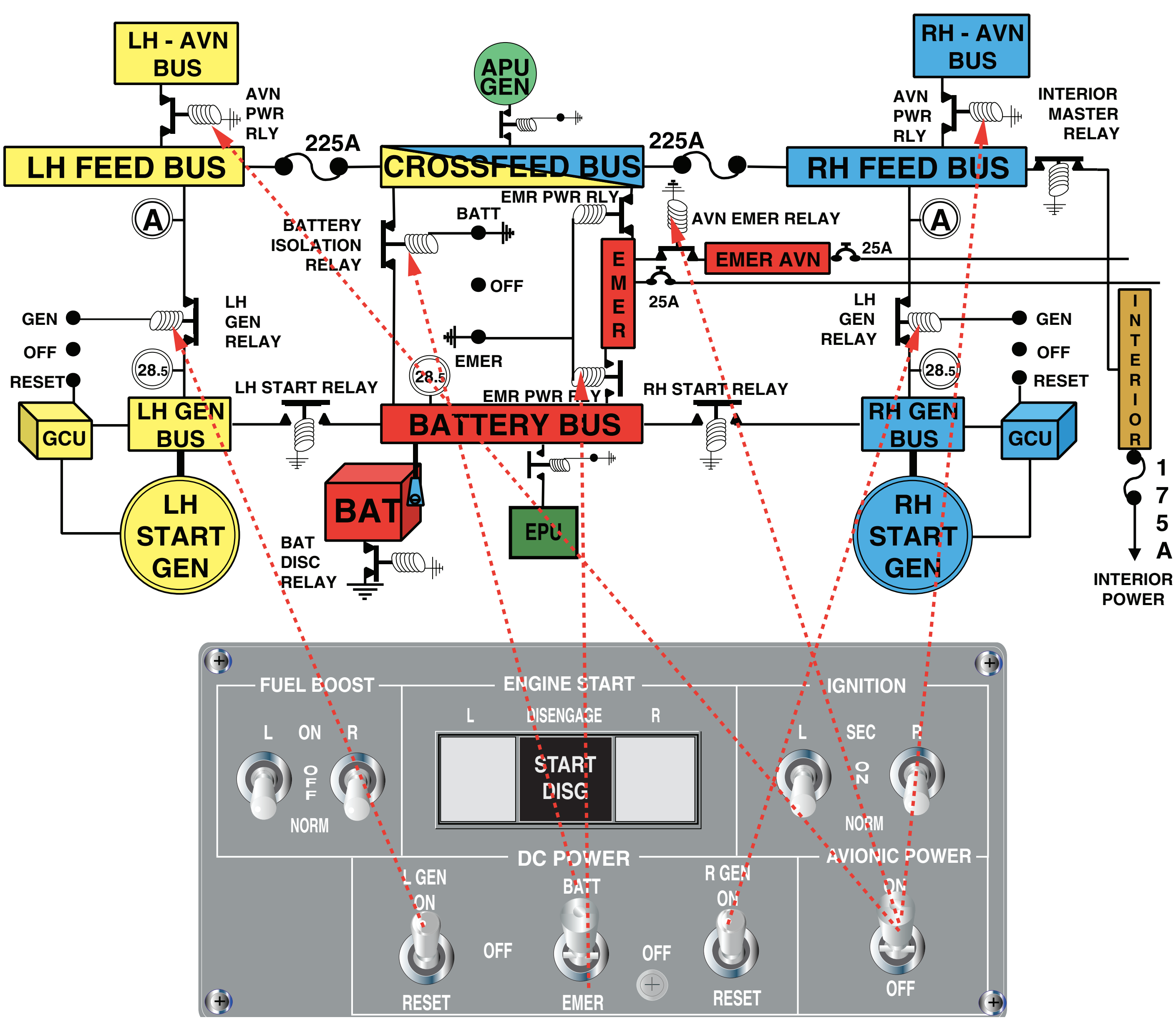


Figure 2-4 Electrical Switch Panel

All major electrical busses are located in the aft J-box, mounted in the tailcone equipment area (Figure 2-5). A diagram on the cover of the J-box displays a map of the bus system, associated relays, current limiters and circuit breakers.

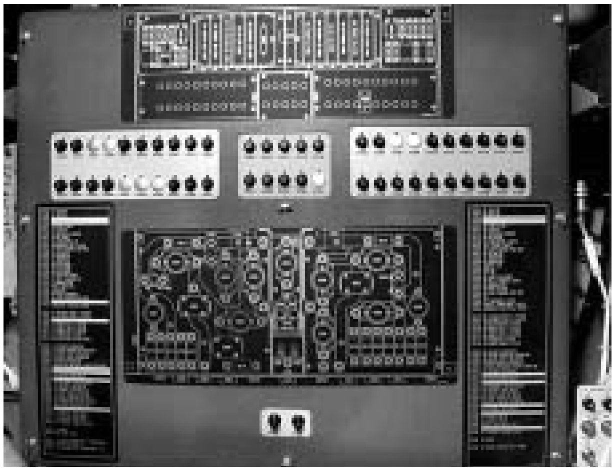


Figure 2-5 Aft Power J-Box

Generators — With the engines running, the generators may be placed on-line by selecting the L and R GEN switches ON (Figure 2-4). This action allows the GCUs to close the associated LH and RH GEN RELAYS and connect power from the generator busses to the LH and/or RH FEED BUSSES. The generator(s) will now power the entire electrical system including charging the battery (BATT switch ON).

BATTERY DISCONNECT SWITCH

Immediately forward of the pilot's circuit breaker panel is a red guarded BATTERY DISCONNECT switch (Figure 2-6). This switch provides the crew with the capability of disconnecting the battery electrically by opening the battery disconnect relay and removing the ground. (Figure 2-3.)

NOTE

The Battery Disconnect Switch will operate only if the Battery Switch is ON (BATT position).

MASTER INTERIOR SWITCH

A Master Interior switch located directly below the Battery Disconnect switch is used to secure all electrical power in the cabin, (Figure 2-6). This switch is normally activated if an electrical fire should occur in the cabin. Placing the switch OFF (UP position), will open the Interior Master Relay on the RH Feed Bus (Figure 2-3), thereby, removing electrical power to the cabin area.

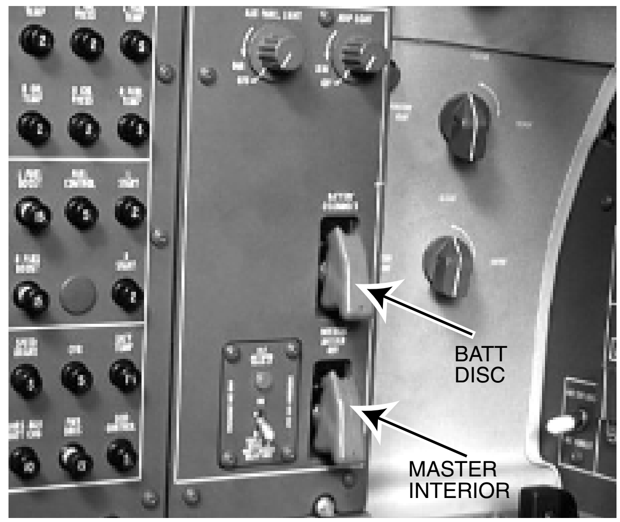


Figure 2-6 Battery Disconnect/ Interior Master Switches

COCKPIT CIRCUIT BREAKER PANELS

The electrical power distribution system is designed to permit logical grouping of circuit breakers in the cockpit.

The Left Hand (LH) Circuit Breaker (CB) panel on the pilot's side wall incorporates CBs for the primary aircraft systems, i.e., ENVIRONMENTAL, LIGHTS, ENGINE INSTRUMENTS, etc., including some basic flight instruments (Figure 2-7).

The RH CB panel on the copilot's side wall incorporates CBs associated with the avionics systems (Figure 2-8).

All FSI training material that incorporates electrical busses and cockpit circuit breaker panels

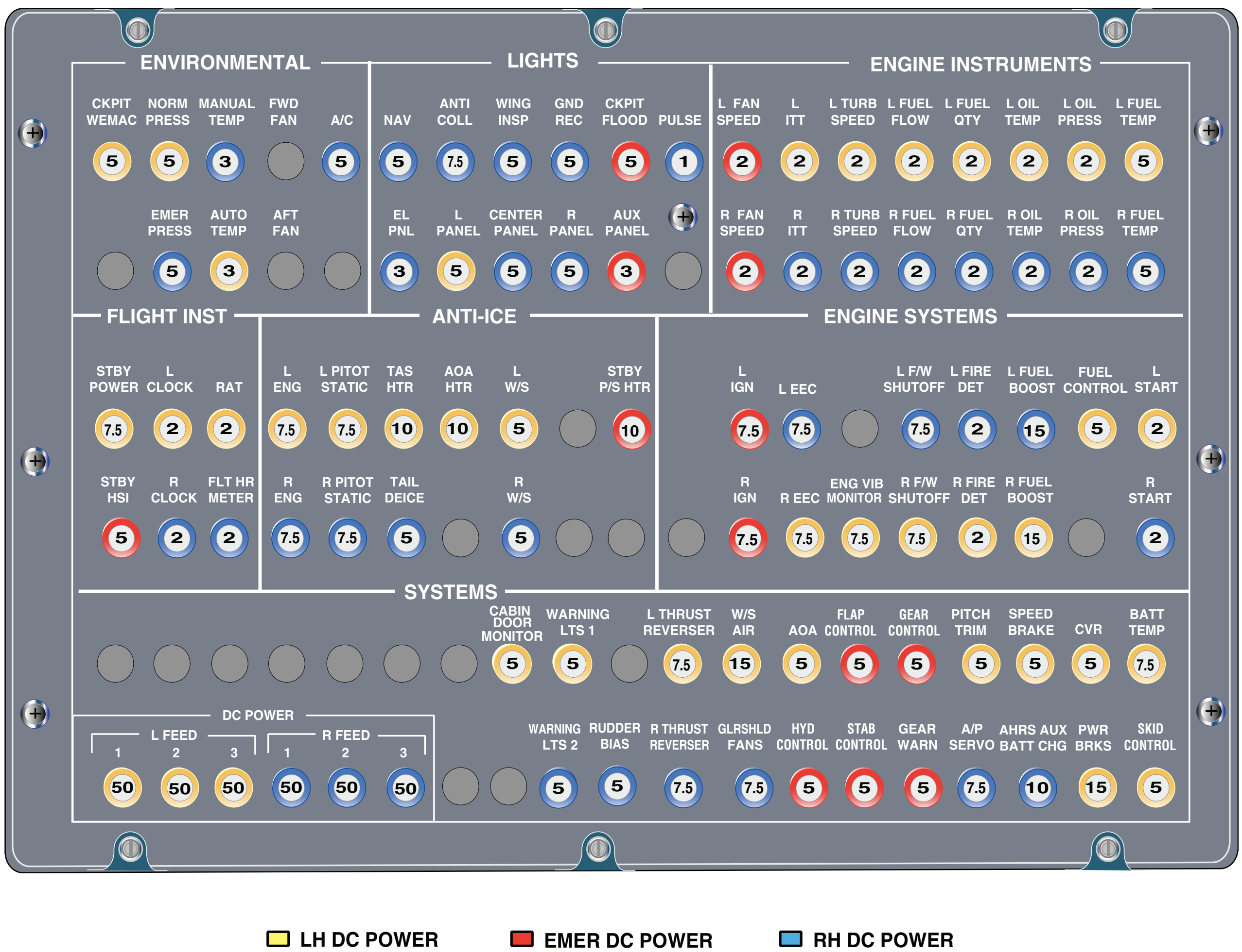
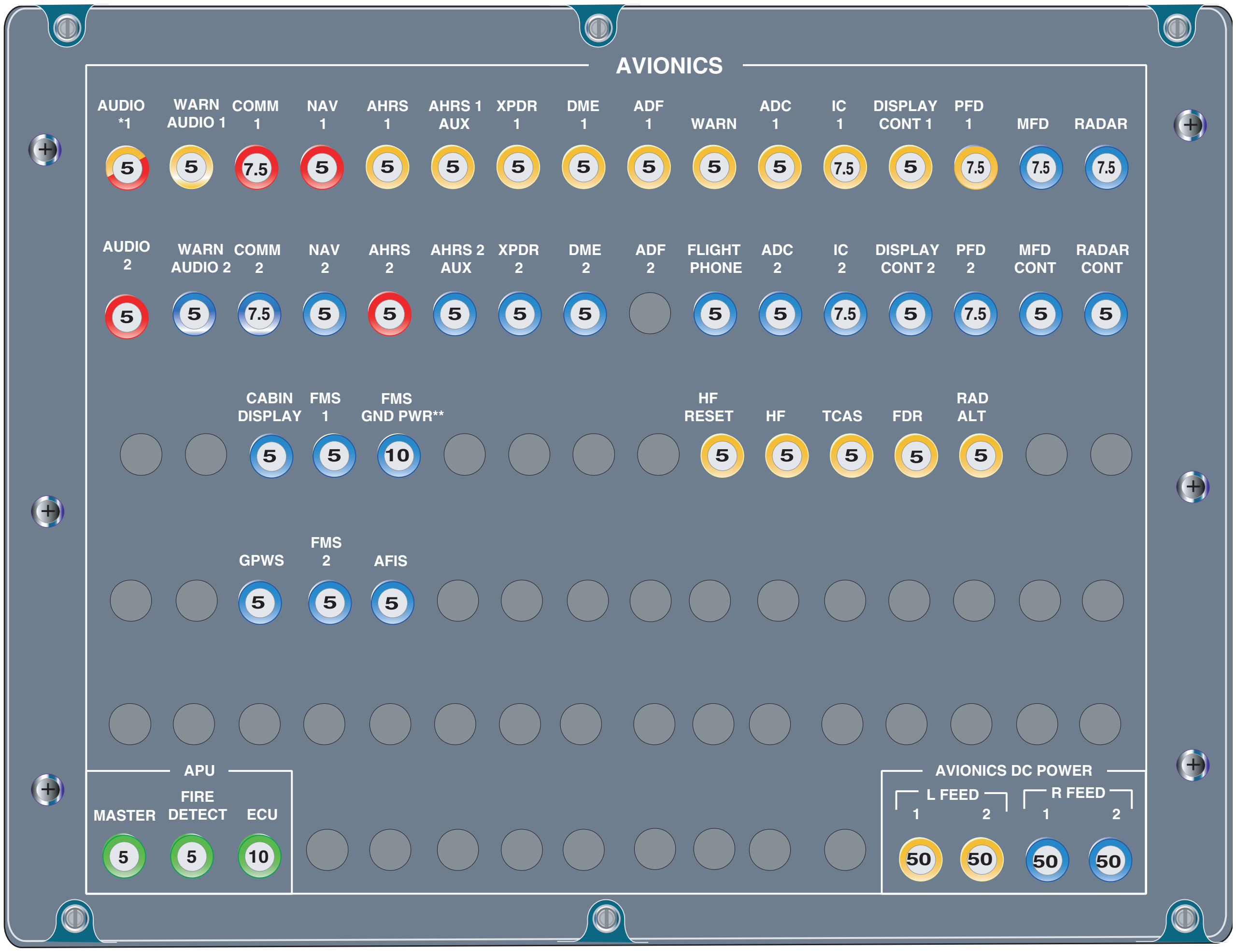


Figure 2-7 Pilot's Circuit Breaker Panel (LH)



LH DC POWER   EMER DC POWER   RH DC POWER

* Normally powered from the LH avionics bus. Switches to the AVN emergency bus with the battery switch in EMER.

** Switches between AHRS AUX PWR Bus and LH AVN Bus based on battery switch and avionics switch position.

NOTE: Circuit breaks in the aircraft are not color coded.

The buses and circuit breakers are coded as follows:

LH Bus Systems	—	Yellow
RH Bus Systems	—	Blue
Emer Bus Systems	—	Red
APU Bus System	—	Green

Figure 2-8 Copilot's Circuit Breaker Panel (RH)

will be color coded to identify LH, RH, EMER and APU bus systems, and associated circuit breakers.

MONITORING

The electrical system is monitored by a voltmeter and ammeters located on the LH instrument panel above the pilot's switch panel (Figure 2-9), annunciator lights, and a battery temperature gage on the RH instrument panel (Figure 2-10).

BATTERY

Battery voltage may be checked with the voltmeter, however the VOLTAGE SEL switch must be in the BATT (spring-loaded) position and the battery isolated from the generators. The voltmeter is connected to the BATTERY BUS with the VOLTAGE SEL in the BATT position.

Battery voltage is checked by placing the BATT switch to either ON or EMER with the generators off-line. If the generators are on- line, the BATT switch is placed to EMER only to check battery voltage.

NOTE

The voltmeter will not register voltage with the BATT switch OFF. the circuit between the BATTERY BUS and the voltmeter is "open" to prevent draining the battery if the aircraft is parked for an extended period with the battery connected.

A **BATT O'HEAT** "red" annunciator segment "flashing" indicates the battery temperature is above 145°F and will trigger both MASTER WARNING RESET lights. If the battery temperature exceeds 160°F, both segments, **BATT O'HEAT** and **>160°**, will flash and trigger the MASTER WARNING RESET lights again, if they had previously been reset.

Battery temperature is monitored by the battery temperature gauge located on the copilot's lower right instrument panel (Figure 2-10).

NOTE

The BATT O'HEAT annunciator and the battery temperature gauge are connected to separate sensors in the battery case.

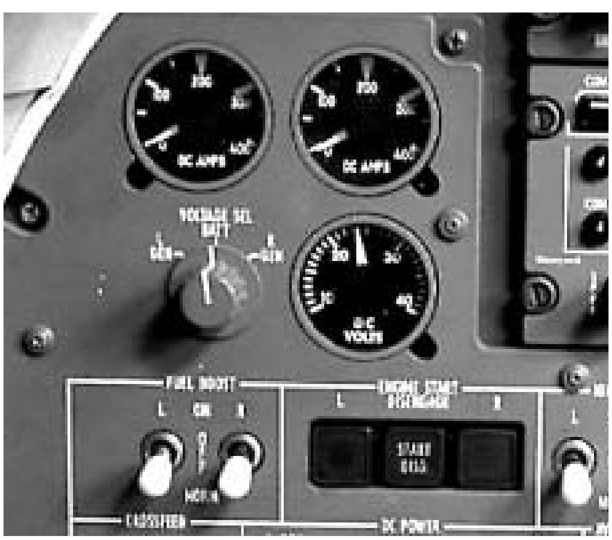


Figure 2-9 Voltmeter and Amperage Gages

Figure 2-10 Battery Temperature Gage

GENERATORS

The generators are monitored by **GEN OFF L** and/or **R** annunciators. They illuminate to indicate the respective generator relay is open and the generator(s) are off-line.

If one generator trips off-line, the respective **GEN OFF L** or **R** annunciator will flash and trigger the master caution lights "steady".

If both generators should trip OFF, both **GEN OFF L/R** annunciators will "flash", and both sets of master caution and MASTER WARNING lights will illuminate.

The voltage selector switch permits monitoring of each generator bus (Figure 2-9).

Placing the voltage selector to L GEN or R GEN will indicate voltage on the selected generator bus. If both generators are on line , assuming the 225 CLs are intact, the voltmeter will indicate system voltage (highest generator voltage). To acquire an accurate voltage check of an individual generator, turn a generator OFF and select the off-line generator and obtain a no-load voltage check. Turning off a generator with the battery switch in BATT and the voltage selector remaining in the BATT position, the voltmeter will indicate the voltage of the opposite on-line generator under high load conditions.

If a generator should trip off-line, the voltage selector and voltmeter may be used to check the affected generator:

- VOLTAGE SEL to affected GEN.
- Obtain voltage reading and:
  - if **"zero"**, attempt RESET and ON with GEN switch (field relay tripped).
  - if **"28.5 or lower"**, reset not probable, GEN RELAY **only opened** to protect the generator.

The ammeters indicate current flow from the generator buses to their respective feed buses. During normal operations with both generators on line, the ammeters should read approximately equal (within 10% of total load). Turning off one generator, the opposite generator amperage load should double (both 225 amp CLs intact).

225 AMP CURRENT LIMITERS/AFT J-BOX CIRCUIT BREAKERS.

The **AFT J-BOX LMT** and **CB** annunciator segments monitor the 225 amp current limiters (CL) between the feed buses and the crossfeed bus (**LMT**), and the left and right start CNTL PCB circuit breakers on the aft power junction box (**CB**).

If a 225 amp CL sensor indicates an open circuit (blown fuse), the **LMT** annunciator segment will illuminate "flashing" and trigger the MASTER CAUTION RESET lights "steady".

If either or both start CNTL PCB circuit breaker(s) on the aft J-Box is open, the **CB** segment will illuminate flashing and trigger the MASTER CAUTION RESET lights "steady".

NOTE

The engine(s) will not start with the CB annunciator illuminated.

PROTECTION

GENERATORS

Generator Control Units (GCU) protect the generators and will trip the generators off line in event an over voltage or under voltage condition exists. Over voltage will trip the generator field relay to prevent damage to the electrical system and other power sources. An under voltage will trip the generator power relay to prevent reverse current to the low voltage generator. If an over voltage should occur and the field relay trips, the generator relay on the same side will open and illuminate the respective **GEN OFF L** or **R** annunciator.

The generators are cooled by engine fan by-pass air directed from the engine by-pass duct and exhausted overboard into the airstream from an exhaust port located on the lower nacelle.

BATTERY

The battery is cooled in flight by ram air that circulates around the battery inside the battery case. The ram air scoop and vent lines are located in the air stream below and in front of the battery compartment door on the left side of the fuselage.

The battery is equipped with a disconnect relay on the ground side of the battery to electrically position the battery off-line. During engine starts utilizing a GPU, the relay opens automatically to save the battery by allowing the GPU to provide all power required for starting. After each engine start is completed the disconnect relay de-energizes closed and the battery is back on-line (detailed in Chapter 7). A red guarded BATTERY DISCONNECT switch located adjacent to the pilot's CB panel (Figure 2-5) allows the pilot to electrically disconnect the battery from the battery bus if required. The BATTERY DISCONNECT switch will operate anytime the battery switch is in the BATT position.

CAUTION

IF THE BATTERY DISCONNECT SWITCH IS LEFT ON FOR AN EXTENDED PERIOD OF TIME WITH THE BATTERY SWITCH IN BATT, THE BATTERY WILL DISCHARGE THROUGH THE DISCONNECT RELAY. THE RELAY IS HELD OPEN ELECTRICALLY UNLESS THE BATTERY IS MANUALLY DISCONNECTED. PLACING THE BATT SWITCH OFF OR EMER WILL ALLOW THE DISCONNECT RELAY TO DE-ENERGIZE CLOSED.

EXTERNAL POWER — OVER VOLTAGE MONITOR

An Over Voltage Sensor monitors voltage output of an external power source. If the voltage output exceeds approximately 32 VDC, the monitor will cause the external power relay to open and protect the electrical system.

CURRENT LIMITERS AND CIRCUIT BREAKERS

Various current limiters and circuit breakers, previously discussed, provide overload protection for electrical buses and associated wiring.

EMERGENCY BATTERY POWER PACKS

SECONDARY FLIGHT DISPLAY (SDF)-MEGGITT BATTERY PACK

A 28-volt, 2.5-ampere-hour, sealed lead-acid battery pack is installed in the left nose compartment. The battery pack can provide approximately 30 minutes of power for emergency operation of the Secondary Flight Display (SFD Meggitt). The pack is normally charged by the aircraft main DC electrical system through the STBY PWR CB on the pilot's circuit breaker panel. The standby SFD battery pack can be checked for adequate charge during preflight by a STBY PWR ON-OFF-TEST switch located on the pilot's lower switch panel (Figure 2-11).

Normal Operations: Placing the STBY PWR switch ON connects the SFD to its battery pack. The SFD will initialize in 180 seconds. If main DC power is available through the STBY PWR CB on the left CB panel, the STBY battery pack is continually recharged (lights adjacent to the STBY PWR switch remain extinguished). Plac-

Figure 2-11 Standby Power Switch — SFD

ing the STBY PWR switch OFF, all power is removed from the SFD. Placing the switch to TEST should display a green light adjacent to the switch to indicate the battery pack is charged.

Emergency Operations: With the switch ON and operating normally, the lights adjacent (green and amber) to the switch will remain extinguished. However, if a loss of main DC power occurs, an "amber" warning light adjacent to the switch will illuminate to indicate that the secondary flight display (Meggitt) is being powered by its dedicated emergency battery pack. If the battery is holding a charge when power loss occurs, approximately 30-minutes of operating time is available. When the emergency battery pack is supplying power independently, it also provides back lighting for the following instruments: Standby HSI and N1 indicators.

EMERGENCY AVIONICS BATTERY PACK

An identical sealed lead-acid battery pack is installed in the right nose compartment. This pack is used as an emergency power supply for the Attitude Heading Reference Systems (AHRS) if power interruptions occur, provided the STBY PWR (Meggitt) switch is ON.

A white **AHRS AUX PWR L–R** annunciator will illuminate if the emergency battery pack is supplying power directly to either or both AHRS systems. The pack is charged from the main DC system through the AHRS 1/2 AUX circuit breakers located on the copilot's CB panel.

The battery pack is capable of providing approximately 30 minutes of operating power directly to both AHRS systems

NORMAL OPERATION

PREFLIGHT

PRELIMINARY INSPECTION. The battery is checked to ensure it is connected properly.

PRELIMINARY COCKPIT INSPECTION. The battery voltage is checked for a minimum of 24 volts. If volts check less than 24 volts, maintenance is required.

PRELIMINARY COCKPIT INSPECTION. The generator switches are ON for normal engine starts and OFF if an external power unit is to be used for engine starts. The battery switch is placed in EMER momentarily and the following emergency instruments are checked to verify that the battery bus emergency relay closed properly: N1 indicators; RMU 1; Standby HSI; and landing gear indicator lights illuminated.

EXTERIOR INSPECTION. The battery switch is ON during the Hot items/Lights inspection. This check should be expedited and the vapor cycle air conditioner OFF, if an external power unit is not used.

COCKPIT PREPARATION. The battery switch must be ON for a few seconds prior to closing the cabin door to perform a self test on the cabin door locking mechanisms. If the CABIN DOOR annunciator remains illuminated, unlock and relock the door, or cycle the battery switch OFF for a few seconds and then ON.

STARTING ENGINES. If the airplane has been cold soaked at temperatures below -10°C (+14°F) and the engines have not been preheated, the use of external power or warming the battery to -10°C (+14°F) or warmer is recommended. Battery temperature may be checked with the battery temperature gauge. Proper battery warmup may require extended application of heat.

NOTE

After the first engine is started, if a cross generator start is anticipated for the second engine, ensure the operating engine is stabilized at idle power.

If a GPU is used for starting, disconnect the GPU before placing the generators ON to ensure the battery is still connected. If a loss of electrical power occurs when the GPU is disconnected, check the BATTERY DISCONNECT switch OFF (cover closed) and/or physical connection in the battery compartment.

Generators, DC amps and volts check. After the generators are placed ON, a DC volts and amps check is conducted as follows:

1. Turn left generator OFF, leave right generator ON. Check left amperage gauge drops to "zero". Right amperage gauge load "doubles". Check voltmeter 28.5 volts, indicates operating generator maintains proper voltage under load. Place VOLTAGE SEL switch to L GEN, check voltmeter 28.5 volts, indicates off-line generator maintains proper voltage under no load.

2. Turn left generator ON and right generator OFF. Check left amperage gauge increases to total load (same as right amperage gauge, previous load), and right amperage gauge drops to "zero". Check voltmeter 28.5 volts, indicates operating generator maintains proper voltage under load. Place VOLTAGE SEL switch to R GEN, check voltmeter 28.5 volts, indicates off-line generator maintains proper voltage under no load.

3. Left and right generators ON. Check L and R amperage gauges parallel (loads equal within 10% of total load). Check system voltage, 28.5 volts.

NOTE

If a satisfactory volts/amps check is conducted as described above, it ensures the generators and GCUs are operating within limits and the 225 amp current limiters are intact (AFT J-BOX LMT annunciator extinguished).

TAXI, TAKEOFF, CLIMB, CRUISE, DESCENT, APPROACH AND LANDING

The electrical gauges should be scanned to verify the electrical system voltage and amperage are remaining within limits.

Shutdown

As the engines are secured with the throttles, the first annunciator associated with the engine shutdown that should illuminate is the respective **GEN OFF L/R** light. As the engine RPM decreases during shutdown, generator voltage begins to decrease to less than the associated feed bus voltage (normally battery voltage), thereby alerting the GCU that an under voltage condition is occurring, and the GCU then trips the generator relay OPEN to prevent reverse current.

Prior to placing the battery switch OFF, scan the engine instruments and annunciator panel for any OFF flags or unusual annunciator light(s) that may be illuminated. May indicate blown current limiters, circuit breakers open, etc.

EMERGENCY/ ABNORMAL OPERATION

ELECTRICAL FIRE OR SMOKE

If an electrical fire or smoke occurs, the crew should don oxygen masks and select EMER oxygen flow. After the crew masks are on, place the oxygen mask microphone selector to MIC OXY MASK. Placing the microphone switch to

MIC OXY MASK position prior to donning the masks may cause a loud regeneration noise through the overhead speakers.

If smoke conditions warrant, smoke goggles should be donned by the crew and ensure the PRESS SOURCE selector is in the NORM position.

If source of electrical fire can be confirmed, pull the appropriate circuit breaker(s) to isolate the problem.

NOTE

If an electrical fire occurs in the cabin, consideration should be given to activating the INTERIOR MASTER switch. All electrical power to the cabin will be shutoff.

If an electrical fire occurs in the cockpit and the source unknown, place the battery switch EMER and the generator switches OFF to shed main DC power and maintain electrical power on the emergency bus system. If this condition occurs at night, place the FLOOD lights and glareshield (AUX PANEL LIGHT) rheostats to full bright.

The following equipment is operational from the EMERGENCY BUSSES:

RH CB panel:

- AUDIO 1 and 2
- COMM 1 (includes STBY COMM 1)
- NAV 1 (includes STBY NAV 1)
- AHRS 2

LH CB panel:

- COCKPIT FLOOD lights
- AUX PANEL lights
- L and R FAN SPEED
- STBY HSI (NAV 1/AHRS 2)
- STBY P/S HTR
- L and R IGNITION (SEC)
- HYD CONTROL
- FLAP CONTROL
- GEAR CONTROL
- STAB CONTROL
- GEAR WARN

NOTE

With the battery switch in EMER and the generators OFF, a properly charged battery may only supply power for approximately 30 minutes.

The standby flight display (Meggitt) will continue to operate on its own emergency battery pack (amber light ON). The battery pack will provide 5 volt power to emergency instrument lighting, i.e., Standby HSI, and N_1 indicators.

Land as soon as practical (within 30 minutes).

Navigate using the Standby HSI and NAV 1, programmed through RMU 1 or the Standby Radio Control Head.

Communicate using COMM1, programmed through RMU 1 or the Standby Radio Control Head. Both audio panels are operational, and the overhead speakers and headsets are functional.

Airplane attitude, altitude and airspeed are referenced from the Standby Flight Display (Meggitt).

Engine RPM is controlled directly by the throttles mechanically linked to the Fuel Control Unit (FCU). The Electronic Engine Controls (EEC) are inoperative. RPM is referenced by the N1 indicators only.

Automatic and manual temperature control is lost. Pressurization must be controlled manually by using the “cherry picker”.

All flight controls and trim are functional (electrical elevator trim is inoperative).

NOTE

If flight through icing is anticipated, engine and wing anti-ice bleed air systems “fail safe” to the ON mode. Ensure the ANTI-ICE PITOT & STATIC switch is ON and the ignition switches are in the SEC position to provide protection for the standby pitot/static system and the engines.

The RAT gauge is inoperative with loss of main DC power.

If Severity of Smoke Warrants

Initiate EMERGENCY PROCEDURES, Section III, AFM - SMOKE REMOVAL and/or EMERGENCY DESCENT and land as soon as possible.

Cockpit Fire

Remove the fire extinguisher located under the copilot’s seat, and remove the pin to arm the extinguisher. Locate the fire and extinguish as appropriate. Land as soon as possible.

Cabin Fire

Remove the portable fire extinguisher stowed in the aft cabin behind the left rear seat. Extinguish the fire as appropriate. Land as soon as possible.

When Landing is Assured

Landing gear, DOWN (emergency bus powered). Flaps, LAND (emergency bus powered). Minimum airspeed, V_{REF}. During landing, use the emergency pneumatic brake system. Refer to ABNORMAL PROCEDURES, Section III, AFM. Multiply charted landing distance by 1.4.

BATTERY OVERTEMPERATURE (BATT O’TEMP WARNING LIGHT ON)

Note amperage on both gauges, select battery switch to EMER and observe an amperage decrease on both gauges. As the battery isolation relay opens, charging power from the generators is removed and a noticeable drop in amperage should be observed.

A one volt decrease within 30 seconds to 2 minutes should be observed on the voltmeter.

Monitor the battery overheat annunciator and the copilot’s BATT TEMP indicator for any possible change, i.e., **BATT O’TEMP >160°** annunciator reilluminates and/or battery temperature gauge indicates rising temperature over 160°F.

IF VOLT/AMP DECREASE

Place the battery switch, OFF. This action returns the emergency bus system connection to the crossfeed bus, which allows the generators to power the emergency buses. The battery load should be reduced to zero allowing battery heat to begin dissipating.

The voltmeter will not register voltage from the battery bus with the battery switch OFF, however, system voltage can be checked by moving the VOLTAGE SEL switch to either L GEN or R GEN.

IF BATT O’TEMP WARNING LIGHT GOES OUT

The crew may elect to place the battery switch ON, if desired. However, be alert for increasing battery temperature.

IF NO VOLT/AMP DECREASE

Normally indicates the battery relay is stuck closed.

Place the battery switch ON and disconnect the battery using the red guarded BATTERY DISCONNECT switch. The battery switch has to be ON to enable the BATTERY DISCONNECT switch.

Note an amperage decrease on both amperage gauges, if the BATTERY DISCONNECT switch successfully disconnected the battery.

IF BATT O'TEMP WARNING LIGHT DOES NOT GO OUT OR > 160 WARNING LIGHT ILLUMINATES

Land as soon as possible

IF BATT O'TEMP WARNING LIGHT GOES OUT

Close guard on the BATTERY DISCONNECT switch and ensure the battery switch is ON. However, be alert for the battery temperature to commence increasing.

Land as soon as practical.

CAUTION

PROLONGED OPERATION WITH THE BATTERY DISCONNECT SWITCH DISCONNECTED AND THE BATT SWITCH ON WILL GRADUALLY DEPLETE THE BATTERY THROUGH THE BATTERY DISCONNECT RELAY.

AFTER LANDING, REFER TO AIRPLANE MAINTENANCE MANUAL FOR PROPER MAINTENANCE PROCEDURES, AS DAMAGE TO THE BATTERY MAY HAVE OCCURRED.

LOSS OF BOTH GENERATORS (GEN OFF L AND R CAUTION LIGHTS ON AND MASTER WARNING ON)

Place generator switches to RESET and OFF. Check each generator voltage with the VOLTAGE SEL switch.

The generator that indicates near normal voltage (28.5), attempt to place ON.

If Only One Generator Comes On

Reduce electrical load not to exceed 300 amps.

NOTE

If the vapor cycle air conditioner is operating in flight, it will trip off line automatically if one generator fails.

The interior Master Switch, located on the pilot's side panel, may be used to shed all non-essential passenger cabin electrical loads.

If Neither Generator Comes On

Select glareshield and flood lights rheostats full bright (night operations only). Place battery switch EMER to open the battery isolation relay and shed the main DC buses. Battery switch in EMER, allows the battery to power the emergency bus system, thus, providing power for essential flight equipment.

A properly charged battery will supply power for approximately 30 minutes to the following equipment:

RH CB panel:

- AUDIO 1 and 2
- COMM 1 (includes STBY COMM 1)

- NAV 1 (includes STBY NAV 1)
- AHRS 2

LH CB panel:

- COCKPIT FLOOD lights
- AUX PANEL lights
- L and R FAN SPEED
- STBY HSI (NAV 1/AHRS 2)
- STBY P/S HTR
- L and R IGNITION (SEC)
- HYD CONTROL
- FLAP CONTROL
- GEAR CONTROL
- STAB CONTROL
- GEAR WARN

The Standby Flight Display (SDF Meggitt) will continue to operate on its own emergency battery pack (amber light ON). The battery pack also provides 5 volt emergency instrument lighting to the standby HSI and the N1 indicators.

Land as soon as practical (within 30 minutes).

Navigate using the Standby HSI and NAV 1, programmed through RMU 1 or the Standby Radio Control Head.

Communicate using COMM1, programmed through RMU 1 or the Standby Radio Control Head. Both audio panels are operational, and the overhead speakers and headsets are functional.

Airplane attitude, altitude and airspeed are referenced from the Standby Flight Display (Meggitt).

Engine RPM is controlled directly by the throttles mechanically linked to the Fuel Control Unit (FCU). The Electronic Engine Controls (EEC) are inoperative. RPM is referenced by the N1 indicators only.

Automatic and manual temperature control is lost. Pressurization must be controlled manually by using the "cherry picker".

All flight controls and trim are functional (electrical elevator trim is inoperative).

NOTE

> If flight through icing is anticipated, engine and wing anti-ice bleed air systems "fail safe" to the ON mode. Ensure the ANTI-ICE PITOT & STATIC switch is ON and the ignition switches are in the SEC position to provide protection for the standby pitot/static system and the engines.
>
> The RAT gauge is inoperative with loss of main DC power.

Operate the emergency lights as desired. The switch is normally in ARM. During a loss of main DC power the emergency exit lights will illuminate automatically.

NOTE

> Emergency cabin lights battery packs will be depleted in approximately 10 minutes if lights remain on.

When Landing is Assured

Landing gear, DOWN; Flaps, LAND; Emergency lights, As desired; Airspeed, VREF.

Landing - Use emergency brake system. Refer to Abnormal Procedures, WHEEL BRAKE FAILURE. Multiply charted landing distance by 1.4.

SINGLE GENERATOR FAILURE (GEN OFF L OR R CAUTION LIGHT ON)

Attempt to reset the generator and ON. If unable, place GEN switch OFF and decrease electrical load not to exceed 300 amps. Turn the vapor cycle air conditioner OFF or WEMAC BOOST LO or HIGH.

NOTE

The vapor cycle air conditioner will automatically load shed in flight if one generator fails.

AFT J-BOX CURRENT LIMITER OR CIRCUIT BREAKER OPEN (AFT J-BOX LMT OR CB CAUTION LIGHT ON)

Indicates an open 225 amp current limiter or ENG START CB open in the aft junction box.

On Ground

Correct prior to flight

In Flight

Monitor the electrical system. Amperage may not parallel if a 225 amp current limiter is open and generator voltages may vary from 25 to 33 volts.

CAUTION

DO NOT TURN OFF THE GENERATORS, PARTIAL ELECTRICAL SYSTEM FAILURE MAY OCCUR ON THE BUS ASSOCIATED WITH A GENERATOR WHICH IS TURNED OFF.

ALTERNATOR BEARING FAILURE (AC BEARING L OR R ADVISORY LIGHT ON)

Indicates impending alternator bearing failure within approximately 20 hours of operation. Maintenance is required.

LIMITATIONS

STARTER

Three Cycles of Operation:

Three engine starts in 30 minutes with a 90 second rest period between start cycles.

BATTERY

Engine Start Limits:

Three starts in one hour.

NOTE

1. If battery limitation is exceeded, a deep cycle including a capacity check must be accomplished to detect possible cell damage. Refer to Chapter 24 of the Maintenance Manual for procedure.
2. Electrical load is limited to 200 amps per generator during ground operations. Transients up to 250 amps are permissible up to 4 minutes.
3. Generator load limit in flight is 300 amps.
4. Three generator assisted cross starts are equal to one battery start.
5. If the BATT O'TEMP annunciator illuminates during ground operation (except for test), do not take off until after proper maintenance procedures have been accomplished.
6. If an external power unit is used for engine start, no battery cycle is counted.
7. Use of an external power source with voltage in excess of 28 vdc or current in excess of 1000 amps may damage the starter. Minimum 800 amps for start.

QUESTIONS

1. A fully charged battery should supply power to the battery bus and the emergency buses for approximately:
   A. 2 hours
   B. 1 hour
   C. 30 minutes
   D. 10 minutes

2. If the red BATT O'TEMP annunciator light segment "flashes," the Battery switch should be initially placed to _________ to isolate the battery from the generators and obtain a voltage reading.
   A. OFF
   B. EMER
   C. Either A or B
   D. None of the above

3. Generators on-line, battery switch BATT, with the voltmeter selector switch remaining in BATT, the voltmeter gauge will indicate:
   A. Generator system voltage, 28.5 V, from the battery bus
   B. Generator system voltage, 28.5 V, from the crossfeed bus
   C. Battery voltage, 24-25 V, from the battery bus
   D. Battery voltage, 24-25 V, from the crossfeed bus

4. Generators on line, battery switch OFF, the voltmeter gauge will indicate:
   A. 24-24V
   B. 28.5 V
   C. No voltage
   D. None of the above

5. If the GEN OFF L annunciator segment illuminates:
   A. The right generator ammeter gage should indicate double the previous load.
   B. The left generator amperage should drop to zero.
   C. The voltmeter should register "zero" with the voltmeter selector remaining in the BATT position.
   D. Both A and B.

6. If both GEN OFF L and R annunciator segments illuminate simultaneously:
   A. The MASTER CAUTION RESET warning lights will illuminate "steady".
   B. The MASTER WARNING RESET warning lights will illuminate "flashing".
   C. Both A and B.
   D. Only the GEN OFF L and R annunciator segments will "flash".

7. If the AFT J-BOX-LMT annunciator segment illuminates, indicates:
   A. An aft J-Box 60 amp feed bus current limiter is "open".
   B. An aft J-Box 225 amp feed bus current limiter is "open".
   C. The generators should be selected OFF one at a time to determine which limiter is "open".
   D. The airplane should be landed as soon as possible.

8. If both generators have tripped off line and unable to reset, and the battery switch is placed in EMER, the following equipment is "inoperative":
   A. Standby HSI
   B. Normal extension of the landing gear
   C. Ignition
   D. Speedbrakes

9. Airplane configuration same as question 8, select correct choice:
   A. Only the Standby Radio Control Head will be operational for air-to-ground communications.
   B. AHRS 2 supplies heading information to the standby HSI.
   C. AHRS 1 supplies heading information to the standby HSI.
   D. The Secondary Flight Display (Meggitt) may be used for an ILS approach (APR switch depressed).

10. If the battery overtemps and cannot be isolated, battery isolation relay stuck closed:
   A. Turn the battery switch OFF and land as soon as practical.
   B. Activate the BATTERY DISCONNECT switch.
   C. After performing choice B, if the BATT O'TEMP>160° annunciator segment begins to flash, land as soon as possible.
   D. Both B and C.

CHAPTER 3
LIGHTING

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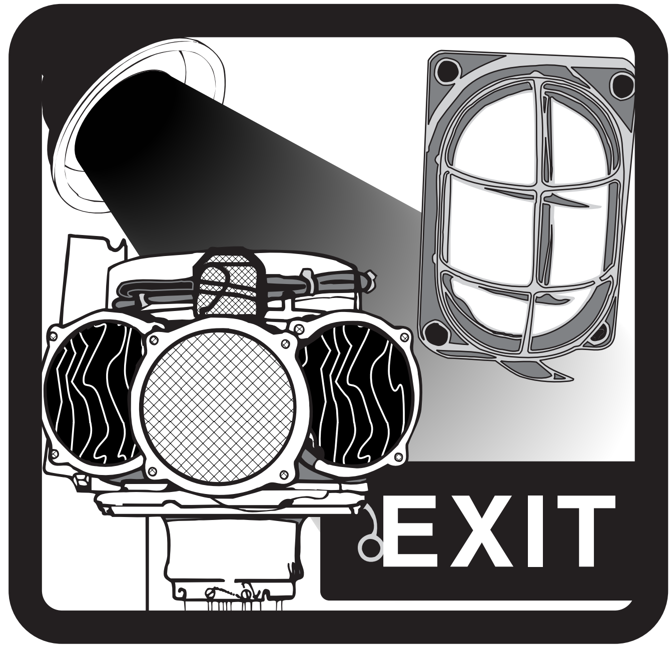
ILLUSTRATIONS

CHAPTER 3
LIGHTING



INTRODUCTION

The Citation EXCEL lighting consists of four major lighting groups, **Interior**, **Emergency**, **Exterior**, and **Tailcone** lighting. All lighting is controlled by switches and rheostats, and protected by separate circuit breakers in the cockpit or in the tailcone J-box.

GENERAL

Interior lighting consists of direct, indirect, fluorescent and incandescent lighting for the cockpit and the cabin.

Emergency lighting consists of a separate and independent system to provide automatic illumination in case of main DC electrical power failure or a +5 G impact.

Exterior lighting consists of direct lights for landing, taxi, recognition, anti-collision, wing inspection, tail flood and a ground recognition beacon.

Tailcone lighting consists of interior lighting in the tailcone and baggage compartment areas.

INTERIOR LIGHTING

Cockpit lighting consists of panel lights that include electroluminescent and backlit instrument lighting, flood lighting, map lights, and glareshield lights.

Cabin lighting consists of overhead fluorescent lights, reading lights for each passenger seat, aft vanity lights, and exit and passenger advisory message lights. Dropped aisle lighting and work station lights are also included.

COCKPIT LIGHTING

Cockpit Panel Lights

Power to the cockpit panel lights are supplied by main DC electrical and controlled by the Master Panel Light ON-OFF (DAY-NIGHT) toggle switch located on the pilot's lower instrument panel (Figure 3-1).

When the master switch is placed ON, the following occurs:

- Master Warning Panel annunciators, dim
- Green ignition lights, dim
- Red ice detect lights mounted on the top forward area of the glareshield illuminate.
- Left, Center, Right, and EL rheostates are active.

The rheostat groups are identified as follows:

LEFT DIM

- LH DIGITAL CLOCK
- AOA INDICATOR
- VOLTMETER
- LH & RH AMP METER
- PILOT'S DISPLAY CONTROLLER
- PILOT'S PFD BEZEL

RIGHT DIM

- OXY GAGE
- RH DIGITAL CLOCK
- BATTERY TEMP INDICATOR
- COPILOT'S PFD DISPLAY CONTROLLER
- COPILOT'S PFD BEZEL
- COCKPIT VOICE RECORDER

Figure 3-1 Cockpit Lights Switch Panel

CENTER DIM

- WET COMPASS
- STDBY HSI
- PRESSURIZATION CONTROLLER
- DIFFERENTIAL PRESS GAGE
- ENGINE GAGES
- FUEL TEMP GAGE
- TRIM TABS
- RUDDER AND AILERON INDICATORS
- RAT INDICATOR
- ECU TEMP INDICATOR
- RADAR CONTROL PANEL
- FMS PANELS

EL DIM*

- THROTTLE QUADRANT
- PILOT'S & COPILOT'S SWITCH PANELS
- LIGHT SWITCH PANELS
- TILT PANEL
- TR ANNU LIGHT/SWITCH PANELS
- LANDING GEAR CONTROL PANEL
- LH & RH CB PANELS
- LH & RH SIDEWALL CB SUBPANELS

** Electroluminescent (EL) panels are used for metal panels with white lettering on a gray background. These panels consist of a layer of phosphor sandwiched between two electrodes and encapsulated between layers of plastic. Electrical power to the EL panels is supplied by a small 40-60 VAC, 400 HZ, inverter located in the nose compartment powered through the EL PANEL CB on the left CB panel.*

OVERHEAD FLOOD LIGHTS

Two cockpit overhead floodlights located near the airplane center line (Figure 3-2) and two engine instrument spot lights mounted in the engine fire tray under the glareshield, wired in parallel to the overhead floodlights, are controlled by the FLOOD DIM rheostat switch located on the pilot's lower instrument (Figure 3-1). The floodlights are powered from the emergency DC bus through the COCKPIT FLOOD CB on the LH circuit breaker panel .

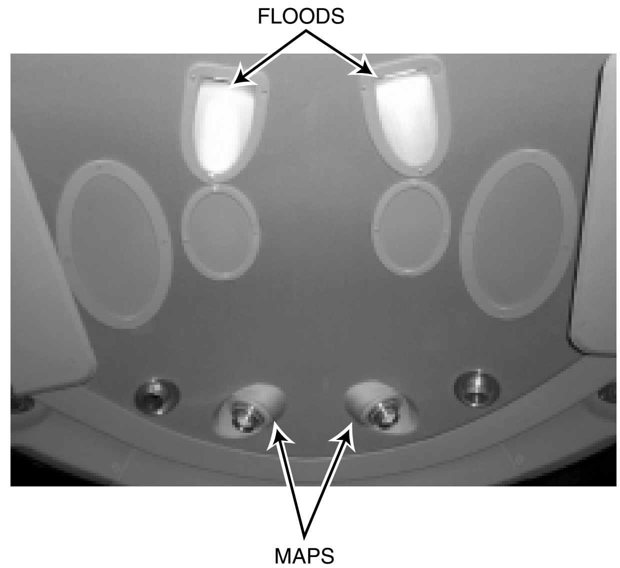


Figure 3-2 Cockpit Flood and Map Lights

OVERHEAD MAP LIGHTS

Two map/chart lights are located in the cockpit overhead liner (Figure 3-2). They can be adjusted by each crew member to aim the lights toward their respective charts without interfering with the other crewmember. The map lights do not require the master ON-OFF switch to be ON. Light intensity is controlled individually by dimming rheostats located forward of each circuit breaker panel and are powered by R PANEL circuit breaker on the pilot's CB panel.

GLARESHIELD

Two fluorescent lamps mounted under the glareshield provide indirect supplemental lighting for the instrument panels. DC electrical power is supplied by a high voltage power supply located in the pilot's side console.

Electrical protection is through the AUX PANEL CB on the left circuit breaker panel powered from the emergency bus. Dimming is controlled by the AUX PANEL LIGHT rheostat on the subpanel forward of the pilot's CB panel.

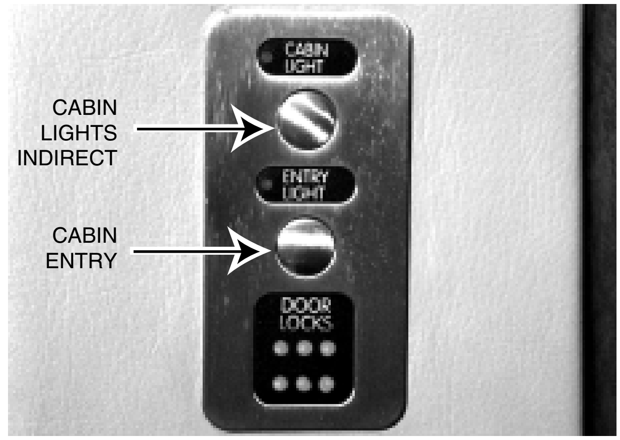
CABIN LIGHTING

OVERHEAD FLUORESCENT LIGHTS

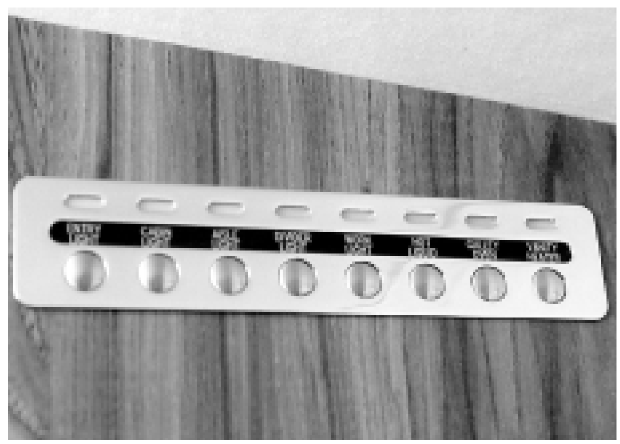
Overhead cabin lighting is provided by dual (upper and lower) fluorescent cold cathode type lamps. The lamps are mounted on both sides of the Passenger Service Unit panels (PSU) providing upper and lower indirect lighting for the cabin (Figure 3-3). This type of lamp is cooler operating, has a longer life and is more shock resistant than conventional fluorescent tubes.

Figure 3-3 Cabin Lights

A cabin light switch panel mounted on the forward cabin entry door frame has a control switch to allow dimming of the cabin indirect lighting, provided main DC power is available (Figure 3-4). Initially pushing the cabin light switch, the indirect lighting will illuminate bright. A few seconds later, pushing the switch again will dim the lights . Pushing the switch again, will extinguish the lights.



FWD ENTRY DOOR FRAME



FWD VANITY SWITCH PANEL

Figure 3-4 Cabin Light Switches

READING LIGHTS

Adjustable overhead reading lights are installed in each passenger service unit above each seat including the aft vanity seat (Figure 3-3). A power switch mounted adjacent to each seat is used to control the lights ON and off anytime main DC electrical power is available.

Adjacent to each reading light switch is a table light switch to illuminate the passenger tables individually.

CABIN ENTRY LIGHTS

An entry light button on the cabin light switch panel mounted on the forward cabin door frame and the forward vanity light switch panel (Figure 3-4) illuminates the following lights, powered from the battery bus:

Five reading lights over cabin seats:

- **Left cabin**: (2 left side, one directly aft of the cabin door and one above the left aft seat);
- **Right cabin**: (3 right side, one above the forward and aft seats, and one above the EMER EXIT door in the aft vanity)

Two Cabin Door Threshold lights, either side of the cabin entry door

Six lights in the cabin door steps.

Figure 3-5 PASS SAFETY Switch, Tilt Panel

AFT VANITY LIGHTS

Various optional vanities may be installed. The components may be individually lighted with power switches installed in each component. The aft vanity overhead lighting is mounted along the top of the cabinet and controlled by a dimming control switch on the vanity switch panel.

The aft closet light is controlled ON and OFF by a door actuated switch.

EXIT AND PASSENGER ADVISORY MESSAGE LIGHTS

Placing the PASS SAFETY switch ON, up position, (Figure 3-5), illuminates all emergency exit lights if the EMER LTS switch is in the ARM position (see EMERGENCY LIGHTS below) and sounds an audible chime and illuminates the following cabin lights:

SEATBELT/NO SMOKING signs, located:

- Right forward cabin /cockpit divider
- Forward side of left aft cabin/ vanity divider
- Left vanity overhead passenger service unit (PSU)

Placing the switch to the SEAT BELT ON, down position, extinguishes the emergency exit lights and the NO SMOKING signs and illuminates the SEATBELT ON signs only, and the chime sounds.

MISCELLANEOUS LIGHTS

A switch panel directly above the forward vanity area, accessible by the copilot, has ON/OFF switches for operating various lights and optional equipment associated with passenger support (Figure 3-4). Pushing the switches ON, illuminates an indicator above each respective switch:

Entry Light — Depressing the ENTRY LIGHT switch ON and OFF controls the entry lights ON and OFF as previously discussed in this chapter under, CABIN ENTRY LIGHTS.

Cabin Light — Depressing the CABIN LIGHT switch ON and OFF controls the cabin lights ON and OFF as previously discussed in this chapter under, OVERHEAD FLUORESCENT LIGHTS

Aisle Light — The dropped aisle lights consist of strip lights attached to the sides of the dropped aisle. The control switch is located on the switch panel above the forward vanity (Figure 3-4).

A portion of the strip lights on each side of the aisle are powered from the EMERGENCY lighting system (see EMERGENCY LIGHTS below).

Divider Light — Some airplanes have a curved tube mounted on the aft divider between the main cabin and the aft vanity area. The control switch is located on the switch panel above the forward vanity (Figure 3-4).

Work Light — A work station light for the forward vanity is controlled by the WORK LIGHT switch on the switch panel above the forward vanity (Figure 3-4).

Miscellaneous Switches — Various control switches associated with optional passenger support equipment may also be mounted on the switch panel above the forward vanity (Figure 3-4).

EMERGENCY LIGHTS

GENERAL

Emergency lighting provide lights for emergency situations. Emergency lights may be powered by main DC Power, or if the main battery is depleted or removed, from emergency nicad battery packs. The cabin emergency lighting system is utilized to provide cabin illumination, emergency exit illumination and identification, emergency egress and ground illumination for emergency evacuation during night time or during conditions of reduced visibility.

DESCRIPTION

The emergency lighting system consists of two nickel-cadmium (nicad) battery packs which are normally charged from the main DC electrical system.

Emergency interior lighting is provided by four selected passenger reading lights, exit identification placards/signs, exit indicators, and dropped aisle lighting. The exit signs incorporate floodlights to illuminate the main entrance and the emergency exit door areas.

Emergency exterior lighting consist of over-the-wing emergency lights mounted in the right fuselage (Figure 3-6) to illuminate the top of the right wing and the area in front of the wing.

OPERATION

The emergency lighting system is operated by a three position EMER LTS (ARM-ON-OFF)

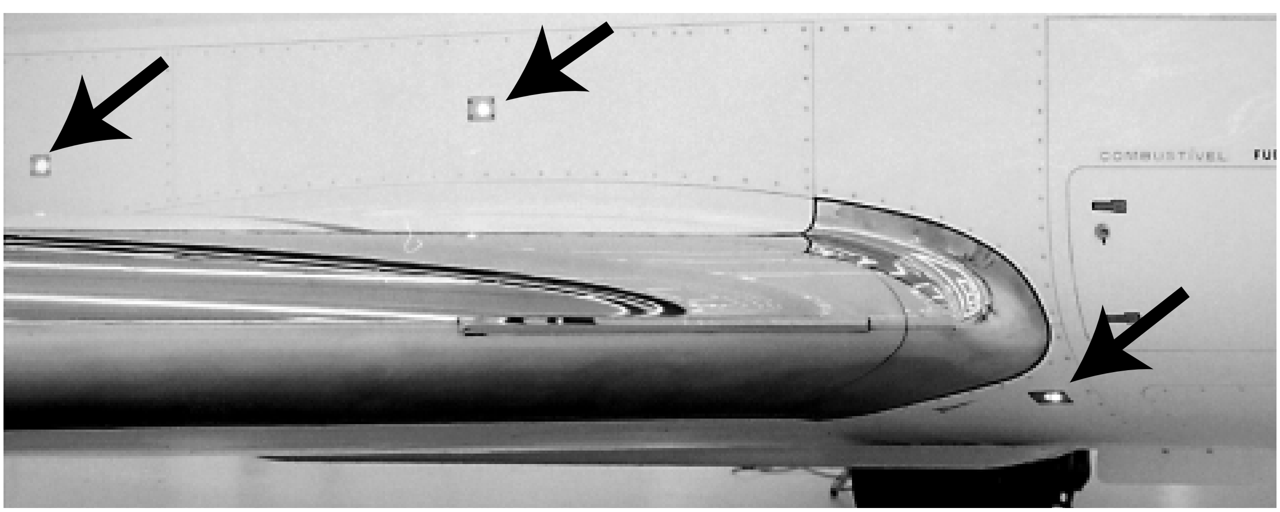


Figure 3-6 Over Wing Emergency Egress Lights

switch located on the pilot's lower switch panel (Figure 3-7). Positioning the switch ON provides power to all emergency lights from the main DC electrical system or from the emergency nicad battery packs in case of total electrical failure.

The EMER LTG switch ARM position provides automatic switching ON to all emergency lights in event total loss of normal DC power occurs or if the aircraft experiences a five gravity (G) force. The EMER LTS switch in the OFF position disables the emergency lighting system.

Adjacent to the EMER LTS switch is an amber light that illuminates when main DC power is on and the EMER LTS switch is OFF (Figure 3-7). This is a reminder for the crew to ARM the system prior to departure.

Cabin Interior Lighting — EMER

Cabin emergency lighting consists of four passenger reading lights which are specially designed. Each of these four reading lamps has two diodes wired between the connecter and the fixture to isolate the normal system from the emergency system.

Placing the EMER LTS switch ON or in the ARM position (if loss of main DC electrical power or a 5-G impact occurs), the following interior lights will illuminate:

Cabin seat lights:

- LH forward
- RH forward
- LH aft
- Over the Emergency Escape Hatch

Cabin exit signs and indicators:

Exit signs,

1. Above main cabin door
2. Above emer escape hatch
3. Left aft cabin/vanity divider (forward face)
4. Every third bulb of the left forward section and right aft section of the dropped aisle lighting.

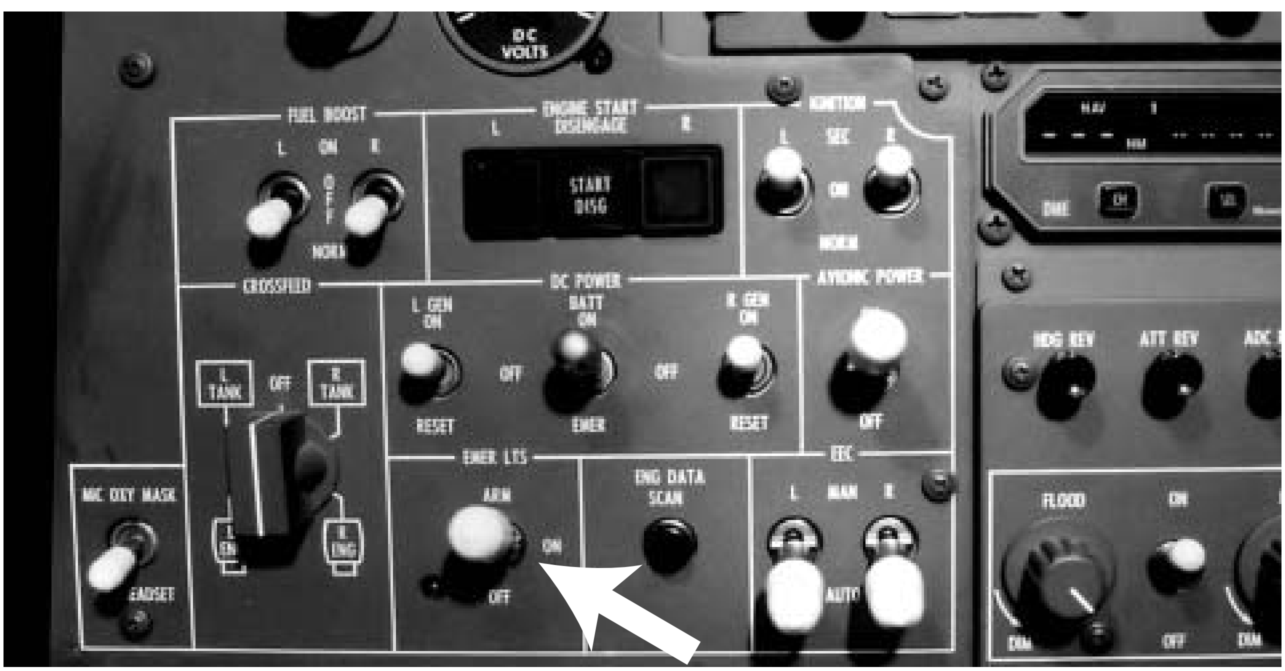


Figure 3-7 Emergency Light Switch, Pilot's Lower Instrument Panel

Exit indicators,

1. Forward of main cabin door
2. Aft of main cabin door
3. Below emer escape hatch

Emergency Exit Sign Lighting

The emergency exit signs are located above the cabin entrance door, on the forward side of the left aft cabin/vanity divider and above the emergency escape hatch, and are internally illuminated when the emergency lighting has been activated .

The EXIT signs above the doors incorporate floodlights designed to illuminate the entire emergency escape hatch and main cabin openings including the first step area of the right wing through the escape hatch and the first step area of the forward cabin entrance door.

Exterior Lighting — EMER

Emergency exterior lighting consist of overwing escape route lights. Two lights mounted in the right fuselage, illuminate the top of the wing, and one light mounted in the fuselage fairing forward of the right wing, illuminates the area in front of the wing.

The lights are powered anytime the emergency lighting system is activated. The aft emergency nicad battery pack will power the lights in event of main DC electrical power failure. (Figure 3-7)

Emergency Lighting Battery Packs

Each battery pack consists of a case, 18 sealed cell rechargeable nicad batteries, various components (relays, blocking diodes, resistors, fuses and circuit breakers) and a five "G" impact/inertia switch. Each pack has a rated capacity of 1.2 ampere hours at a voltage of 22 VDC.

The forward battery pack is mounted in the pilot's left side console and the aft pack is mounted in the aft vanity area. The primary function of the battery packs is to (1) supply electrical power to the emergency lights when main DC power is lost (loss of both generators, battery switch in EMER or (2) if a 5G impact is experienced.

The emergency lighting load is divided between the battery packs to ensure adequate lighting in case of a single battery pack failure.

Forward Battery Pack

The forward battery pack provides power to illuminate the exit indicators on either side of the cabin door, an exit sign over the cabin door, an overhead reading light opposite the cabin door, an overhead reading light aft of the cabin door, and every third bulb in the forward section of the left dropped aisle lights leading to the cabin entry door.

Aft Battery Pack

The aft emergency battery pack provides power to illuminate an exit sign above the rear escape hatch, an exit sign on the forward side of the cabin's aft divider, an overhead light above the escape hatch, an overhead reading light above the left aft seat in the cabin, every third bulb in the aft section of the right dropped aisle lights leading to the emergency escape hatch, and three exterior lights for over wing escape.

EXTERIOR LIGHTING

Exterior lighting consists of navigation lights, anticollision (strobe) lights, ground recognition lights (beacon), wing inspection lights, landing lights, recognition lights, belly fairing lights (taxi), and tail floodlights.

Exterior lights are powered from the main DC electrical system and controlled by switches located on the tilt panel (Figure 3-8) and on the center pedestal (Figure 3-9).

Figure 3-8 Exterior Lighting Switches, Tilt Panel

Figure 3-9 Landing Light Switches

Navigation Lights

Navigation lights consist of a colored light on each wingtip, left wingtip-red, right wingtip-green, and a white (clear) light in the tailcone stinger (Figure 3-10). The lights are controlled by a NAV ON-OFF switch on the LIGHTS sub-panel located on the tilt panel (Figure 3-8).

Anti-Collision Lights

Anti-collision (strobe) lights are mounted on the extreme outboard end of each wingtip. The lights are extremely high intensity pulsating strobes and should not be used on the ground except just prior to takeoff and secured shortly after landing.

The anti-collision lights are controlled by the GND REC/ANTI-COLL switch located on the tilt panel (Figure 3-8). Positioning the switch up to the ON position will illuminate both wingtip anti-collision lights and the tail ground recognition light (red beacon).

Ground Recognition Light

The ground recognition light consists of a red beacon located on top of the rudder. The light is controlled by the GND REC/ANTI-COLL switch located on the tilt panel (Figure 3-8).

Positioning the switch to the center GND REC ON position will illuminate the red tail beacon light only.

Positioning the switch to the upper GND REC/ANTI-COLL ON position will illuminate the wingtip anti-collision strobes simultaneously with the beacon light.

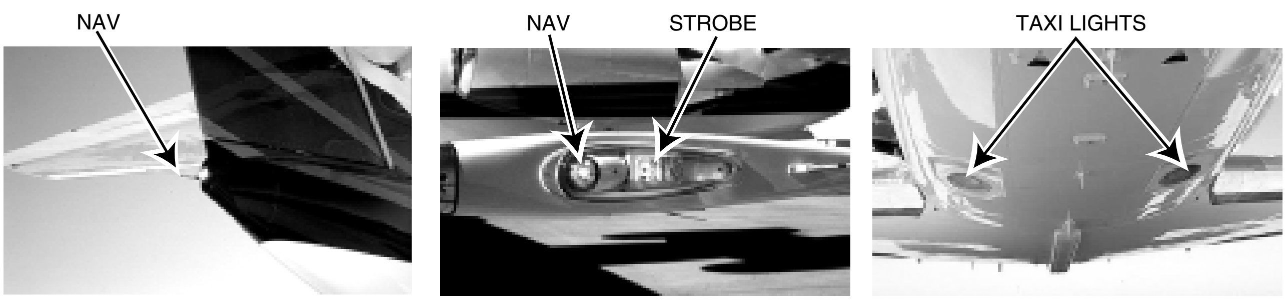
Wing Inspection Lights

Wing inspection lights are located in the left and right sides of the fuselage forward of the wing leading edges. The lights are used to visually check the wing leading edges for ice buildup during night operations. The lights are controlled by a single WING INSP ON-OFF toggle switch located on the tilt panel, ANTI ICE/DEICE sub-switch panel.

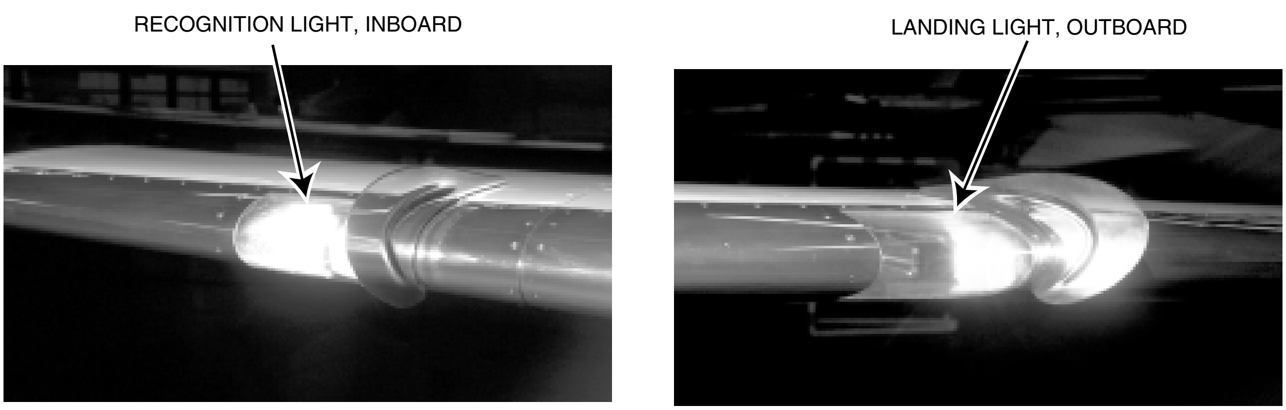
Landing/Taxi Lights

The airplane is equipped with a set of taxi lights. The set is mounted in the belly fuselage fairings on each side of the fuselage and are installed in a fixed position (Figure 3-10). They are seal beam lights controlled by two three position LANDING LIGHTS switches mounted on the center pedestal (Figure 3-9). The belly fuselage taxi lights also function as supplemental landing lights.

TAXI/RECOGNITION/LANDING LIGHTS, ON



NAVIGATION LIGHTS AND FUSELAGE BELLY TAXI LIGHTS, OFF



LANDING AND RECOGNITION LIGHTS, ON

Figure 3-10 Taxi/Recognition/Landing Lights

Landing Lights

Lights mounted in the left and right wingtips serve as landing lights (Figure 3-10). The LANDING lights are located outboard of the wing recognition lights on each forward wingtip in the same light assembly. The lights are canted down slightly.

The lights are controlled by the LANDING LIGHTS switches mounted on the center pedestal (Figure 3-9).

Recognition Lights-Wing Tip

The wingtip recognition lights are located in forward wingtip light assemblies inboard of the LANDING lights (Figure 3-10).

The lights are controlled by the LANDING LIGHTS switches mounted on the center pedestal (Figure 3-9).

NOTE

The LANDING LIGHTS switches on the center pedestal next to the ROTARY TEST selector control the following lights:

<u>REC/TAXI-ON (down position)</u> "BELLY" TAXI LIGHTS and RECOGNITION LIGHTS-ON (four lights)

<u>OFF (center position)</u> All LANDING, TAXI and RECOGNITION LIGHTS-OFF

<u>LANDING LIGHTS-ON (up position)</u> All LANDING, RECOGNITION and BELLY TAXI LIGHTS-ON (six lights)

Tail Floodlights

The tail flood lights, also known as logo or identification lights, are fixed position lights located on the left and right horizontal stabilizers. The lights are canted to illuminate the vertical stabilizer during night operations for identification purposes. They are controlled by the TAIL FLOOD ON-OFF light panel switch located on the tilt panel (Figure 3-8).

TAILCONE COMPARTMENT LIGHTS

The tailcone compartment lights are powered from the battery bus. This eliminates the necessity of entering the cockpit to turn on the tailcone maintenance compartment and/or baggage compartment lights.

TAILCONE MAINTENANCE

The tailcone maintenance compartment light is located above and behind the tailcone access door. The light is controlled by an ON-OFF toggle switch forward of the access door adjacent to the right side of the electrical J-box.

If the light is inadvertently left ON, closing the compartment access door will extinguish the light (microswitch activated by closing the lower rear latch assembly).

TAILCONE BAGGAGE

The tailcone baggage compartment is illuminated by three lights, two overhead ceiling lamps and one sidewall lamp. The lights are all controlled by a manual ON-OFF toggle switch located in the access door closeout.

The manual toggle switch is wired in series with a door frame microswitch. If the lights are inadvertently left ON, the microswitch will extinguish the lights when the door is closed and latched.

OPTIONAL—AUTOMATIC PULSELITE SYSTEM

The Precise Flight, Inc. Automatic Pulselite System provides pulsing of the taxi (belly) and recognition lights. The system is automatically activated when both REC/TAXI LIGHTS switches are selected ON (down) and the airplane is airborne.

Selecting one or both, landing light switches to LANDING LIGHTS ON (up) will deactivate the system (the lights will illuminate steady).

Upon landing, the left squat switch will cause the Pulselite system to be overridden and all lights (taxi/recognition lights) will revert to steady illumination.

The Pulselite system requires main DC power and is protected by a 1amp circuit breaker labeled PULSE on the pilot’s CB panel.

An optional pulselight switch may be installed in the cockpit, normally adjacent to the LANDING/REC/TAXI LIGHTS switch. The switch may be used to override the squat switch to allow pulsing of the taxi/recognition lights on the ground. The switch must be ON in addition to having both REC/TAXI LIGHTS switches ON for the lights to pulse (airborne or on the ground). Selecting one or both LANDING LIGHTS ON, or pulselight switch OFF, will deactivate the system.

Refer to SUPPLEMENT 2, PRECISE FLIGHT - AUTOMATIC PULSELIGHT SYSTEM in the Airplane Flight Manual (*AFM*) for operating procedures.

QUESTIONS

1. Turning the PANEL LIGHT CONTROL master switch to ON:
   A. Activates the control rheostats
   B. Dims the annunciator panel lights
   C. Illuminates the STARTER DISENGAGE button.
   D. All of the above

2. Emergency cabin lighting is powered from:
   A. Main aircraft battery.
   B. Two emergency battery packs.
   C. Emergency DC power.
   D. Either A or B.

3. Emergency lighting is activated by:
   A. Flood light switch.
   B. Emergency light switch ON.
   C. Loss of main DC power or 5g force (switch ARM).
   D. B and C.

4. Landing lights consist of:
   A. Belly lights only.
   B. Belly lights and recognition lights.
   C. Both wing tip lights on each wing tip.
   D. Outboard wing tip lights.

5. Which lights will remain working when the Battery switch is placed to EMER (generators off-line)?
   A. Floods and Auxiliary Panel lights.
   B. Flood and Map lights
   C. EL and Auxiliary Panel lights.
   D. Left, Right, and Center Panel lights.

CHAPTER 4
MASTER WARNING SYSTEMS

CONTENTS

ILLUSTRATIONS

TABLES

CHAPTER 4
MASTER WARNING SYSTEM

INTRODUCTION

The master warning system on the EXCEL provides warning of airplane equipment malfunctions. It provides indications of an unsafe operating condition requiring immediate attention, crew advisory warnings that require attention but not necessarily immediate action, and advisory indications that some specific systems are in operation.

GENERAL

The master warning and master caution annunciator panel light system consists of two master warning light switches and two master caution light switches, and an annunciator panel light cluster which provide visual indications to the flight crew of certain conditions and/or functions of selected systems. Each annunciator segment has a legend which illuminates to indicate an individual system fault or advisory.

The entire warning light system requires main DC electrical power. The system is protected by WARNING LTS 1 (LH bus system) and WARNING LTS 2 (RH bus system) circuit breakers on the pilot's CB panel.

MASTER WARNING RESET SWITCH LIGHTS (RED)

Two "red" MASTER WARNING RESET lights are located on the instrument panel, one each on the pilot's and copilot's instrument panels. When any "red" light illuminates "flashing" on the annunciator panel, both master warning lights illuminate "flashing" simultaneously until reset by depressing either MASTER WARNING RESET switch light, which will extinguish both master warning lights. Resetting the master warning lights will result in the "red" flashing annunciator light to revert to "steady" illumination until the malfunction is cleared.

If both "amber" GEN OFF L and R annunciators are flashing simultaneously the MASTER WARNING lights will commence flashing.

NOTE

If a partial loss of main DC power occurs to a MASTER WARNING RESET indicator, i.e., a WARNING LTS 1 or 2 circuit breaker open (LH CB panel), or a main 225-amp current limiter blown and the generator is off line on the same side, the opposite master warning indicator will illuminate "steady."

If a master warning input from the annunciator panel subsequently occurs, the "steady" illumination of the affected master warning light will change to flashing. Resetting the affected master warning light will cause the light to return to "steady" illumination.

MASTER CAUTION SWITCH LIGHTS (AMBER)

There are two MASTER CAUTION RESET lights located on the instrument panel, one each located adjacent to each MASTER WARNING RESET light. These lights will illuminate "steady" when any "amber" annunciator panel light flashes.

NOTE

There are exceptions when a "flashing" white light will trigger the MASTER CAUTION RESET lights

The various amber annunciators may initially illuminate "steady" for a few seconds, or may initially "flash." Lights that initially illuminate "steady" for a few seconds are normally lights that may illuminate for a short period of time during normal system operations. The short time delay for these lights vary from light to light before they begin to flash and illuminate the MASTER CAUTION RESET lights. The system is designed to prevent nuisance illuminations of the master caution lights.

Resetting the MASTER CAUTION RESET lights by depressing either switch light will extinguish both master caution lights and cause the "amber" annunciator panel light to illuminate "steady" until the malfunction is cleared.

NOTE

The MASTER CAUTION RESET and MASTER WARNING RESET lights are inhibited from illuminating during initial electrical power-up (Battery Switch — BATT ON). All normal active inputs to the annunciator panel at time of power-up will cause those associated lights to illuminate "steady" to prevent the master caution and master warning lights from illuminating.

ANNUNCIATOR PANEL

The annunciator panel is mounted on the center portion of the glareshield above the center instrument panel and contains a cluster of "red" warning, "amber" caution, and "white" advisory lights. The annunciator lights operate in conjunction with MASTER WARNING RESET

and MASTER CAUTION RESET lights. When a malfunction is detected, the associated annunciator illuminates until the malfunction is cleared. Burned out bulbs can be replaced by pushing in the light assemblies and using a tool to remove the assembly with the burned out bulb.

Annunciators lights are classified as WARNING, CAUTION, and ADVISORY. Warning lights are generally "red" (except failure of both DC generators). Red lights indicate a warning malfunction which requires immediate corrective action. The red warning lights in the annunciator panel will cause the "red" MASTER WARNING RESET lights to "flash." Failure of both generators (GEN OFF L and R both "flashing") triggers both the MASTER WARNING RESET and MASTER CAUTION RESET lights.

NOTE

Illumination of the LH and/or RH ENGINE FIRE light(s) will not trigger the MASTER WARNING RESET lights. The FIRE lights are direct-functioning system lights and are not part of the airplane annunciator indicating system.

WARNING LIGHTS (RED)

When a "red" WARNING light on the annunciator panel illuminates, it will immediately illuminate "flashing" and cause the red MASTER WARNING RESET lights to illuminate "flashing." Acknowledging the malfunction by resetting either MASTER WARNING RESET light will result in extinguishing both MASTER WARNING lights and the red annunciator panel light(s) will change from "flashing" to "steady."

If the malfunction is cleared prior to resetting the master warning lights, the red annunciator panel light will extinguish but the red MASTER WARNING RESET lights will continue to "flash" until they are reset.

CAUTION LIGHTS (AMBER)

Annunciator panel CAUTION lights are "amber." Amber lights indicate either a malfunction that requires immediate attention, but not necessarily immediate action, or abnormal system operation.

The amber lights, located in the annunciator panel, come on either "steady" momentarily and then commence "flashing" or may commence "flashing" immediately. When an amber annunciator panel light "flashes," the MASTER CAUTION RESET lights illuminate "steady." Resetting either master caution light will cause the respective "amber" annunciator light(s) to illuminate "steady" until the abnormal condition is cleared.

If the condition is cleared prior to resetting the master caution lights, both MASTER CAUTION RESET lights and the annunciator will extinguish prior to attempting a reset.

ADVISORY LIGHTS (WHITE)

Various ADVISORY lights located on the annunciator panel are identified "white" and only illuminate "steady," with two exceptions, GND IDLE and FUEL XFEED (Table 4-1). When these lights illuminate they do not normally cause the MASTER CAUTION RESET lights to illuminate, except as noted above. The advisory lights are not critical to safety-of-flight operations and normally indicate either routine system operations or minor abnormal situations.

Consult the ABNORMAL PROCEDURES CHECKLIST or the ABNORMAL PROCEDURES section in the *AFM* or *XL PTM, Volume I*, for any possible corrective action required or advisory information which may require systems monitoring.

ILLUMINATION CAUSES

Table 4-1 illustrates each annunciator light placard, color, and cause for illumination.

Table 4-1 ANNUNCIATOR ILLUMINATION CAUSES

ANNUNCATOR	CAUSE FOR ILLUMINATION	ANNUNCATOR	CAUSE FOR ILLUMINATION
BATT O'TEMP / >160	**BATTERY OVER TEMP** Flashes if battery temperature is >145°F. Activates MASTER WARNING lights. If battery temperature increases >160°F, entire light element commences to flash, activates MASTER WARNING lights.	STAB MIS COMP	**STABILIZER MISCOMPARE** Annunciator illuminates flashing to indicate that the horizontal stabilizer is not in agreement with the flap handle position after 30 seconds travel time. Activates MASTER CAUTION lights. On the ground, the NO TAKEOFF annunciator will also illuminate steady.
CAB ALT / ------	**CABIN ALTITUDE** Flashes if cabin altitude is >10,000 ft during normal operations. During operations at airfields above 8,000 feet and aircraft is below FL 245, the annunciator flashes if the cabin altitude is >14,500 feet. Activates MASTER WARNING lights.	SPD BRK EXTEND	**SPEED BRAKE EXTENDED** Annunciator illuminates steady to indicate both speed brakes are fully extended. On the ground, the NO TAKEOFF annunciator will also illuminate.
LO OIL PRESS / L R	**L/R LOW OIL PRESSURE** Annunciator flashes if the respective engine oil pressure is <20 psid. Activates MASTER WARNING lights.	ENG VIB / L R	**L/R ENGINE VIBRATION** Annunciator illuminates steady to indicate the respective engine vibration monitor has detected engine vibration beyond prescribed limits.
LO HYD FLOW / L R	**L/R LOW HYDRAULIC FLOW** Annunciator illuminates steady to advise the crew that L and/or R engine-driven hydraulic pump flow rate is below normal. After five seconds it will begin flashing and illuminate MASTER CAUTION lights.	OIL FLTR BP / L R	**L/R OIL FLTER BYPASS** Annunciator flashes to indicate the respective engine oil filter bypass switch has activated to indicate an impending oil bypass condition. Activates MASTER CAUTION lights.
LO HYD LEVEL	**LOW HYD LEVEL** Annunciator flashes to indicate that the hydraulic reservoir level is low. Activates MASTER CAUTION lights.	GND IDLE	**LOW SPEED GROUND IDLE.** ON GROUND; normal annunciation with EECs in AUTO mode. IN FLIGHT — if L and/or R EEC reverts to ground idle mode, GND IDLE flashes and activates MASTER CAUTION lights. Annunciator illuminates steady with no illumination of master caution eight seconds after the airplane transitions from flight to ground with the EECs in auto mode.
HYD PRESS	**HYDRAULIC PRESSURE ON** ON GROUND – Annunciator illuminates steady with no illumination of master caution to indicate the hydraulic system is pressurized. IN FLIGHT – Annunciator illuminates steady with no illumination of master caution to indicate the hydraulic system is pressurized. If still on after 40 seconds, annunciator begins to flash and activates MASTER CAUTION lights.	NO TAKEOFF	**NO TAKEOFF** ON GROUND, illuminates steady to indicate one or more of the following: flaps are <7° or >15°, elevator is out of trim for takeoff, horizontal stabilizer is out of the takeoff position (STAB MISCOMP), and/or the speed brakes are not completely stowed. Advancing power beyond approximately 80% N_1 with any of the above conditions existing, will activate the MASTER CAUTION lights and an aural warning sound.

Table 4-1 ANNUNCIATOR ILLUMINATION CAUSES (cont.)

ANNUNCATOR	CAUSE FOR ILLUMINATION	ANNUNCATOR	CAUSE FOR ILLUMINATION
P / S HTR L R	**L/R PITOT/STATIC HEATER** ON GROUND — annunciator illuminates steady to indicate the pitot-static heater switch is OFF. IN FLIGHT — annunciator flashes to indicate the switch is OFF or an inoperative heating element, activates MASTER CAUTION lights.	ENG ANTI - ICE L R	**L/R ENG ANTI-ICE** ON GROUND, annunciator illuminates steady to indicate the nacelle lip temperature is low. The annunciator will extinguish when normal operating temperature is reached. IN FLIGHT, turning on the engine anti-ice system, the annunciator initially illuminates steady, and if engine anti-ice has not reached normal operating temperature within 4 min, 45 sec, the light flashes and activates the MASTER CAUTION lights. Placing the engine anti-ice switch OFF, if the fan stator valve remains “open,” the annunciator will flash and activate the MASTER CAUTION lights.
EMER PRESS	**EMERGENCY PRESSURIZATION** Annunciator flashes to indicate emergency pressurization has been manually selected or automatically activated by an air cycle machine overheat, NORM PRESS CB open, or the cabin altitude is above l4,500 feet, activates MASTER CAUTION lights.		
ACM O'HEAT	**ACM OVERHEAT** Annunciator flashes to indicate the ACM has overheated and automatically shut down. EMER PRESS automatically activates, activates MASTER CAUTION lights.	FUEL GAUGE L R	**L/R FUEL GAUGE** Annunciator flashes to indicate the respective fuel quantity gauging system has detected a fault. Activates MASTER CAUTION lights. Note: Record the signal generator BITE indications prior to securing electrical power.
AP PITCH MISTRIM	**AP PITCH MISTRIM** Annunciator flashes to indicate the autopilot elevator servo is not trimmed properly, (excessive sustained pressure). UP or DN light will illuminate on the A/P controller, activates MASTER CAUTION lights.	LO FUEL LEVEL L R	**L/R LOW FUEL LEVEL** Annunciator flashes to indicate low fuel quantity in the respective tank. (360 lbs $\pm$ 20 lbs). Measured by a float switch, activates MASTER CAUTION lights.
AP ROLL MISTRIM	**AP ROLL MISTRIM** Annunciator flashes to indicate the autopilot aileron servo is not trimmed properly (excessive sustained pressure), activates MASTER CAUTION lights.	EEC MANUAL L R	**EEC MANUAL** Advisory — Electronic Engine Control is off-line (failed or selected off) and the engine is operating in manual mode. Throttle detents and engine sync are inoperative.
AHRS AUX POWER 1 2	**1/2 AHRS AUXILIARY POWER** #1 and/or #2 annunciator illuminates steady if the respective AHRS has reverted to the AUX battery pack due to a loss of normal DC power, i.e., (Avionics switch OFF).	GEN OFF L R	**L/R GEN OFF** Annunciator flashes to indicate the respective Generator Relay is open and the generator is off line. Activates MASTER CAUTION lights. Both GEN OFF L/R flashing simultaneously will activate both MASTER WARNING and MASTER CAUTION lights.

Table 4-1 ANNUNCIATOR ILLUMINATION CAUSES (cont.)

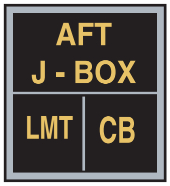
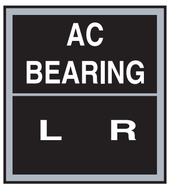
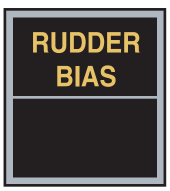
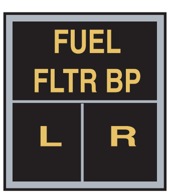
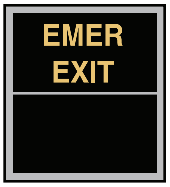
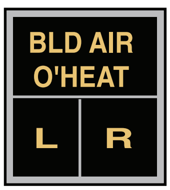
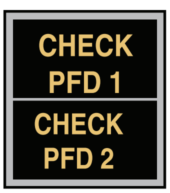
ANNUNCATOR	CAUSE FOR ILLUMINATION	ANNUNCATOR	CAUSE FOR ILLUMINATION
	**LMT/ CB AFT J-BOX** LMT annunciator flashes if L and/or R 225-amp crossfeed bus fuse limiter (located in the aft J-box) is "open." CB annunciator flashes if L and/or R start control PCB circuit breaker(s) located in the aft J-box is/are open, disabling the engine(s) start system. Activates MASTER CAUTION lights.		**ANTISKID INOP** ON GROUND, annunciator flashes if the antiskid system is inoperative, activates MASTER CAUTION lights. IN FLIGHT, annunciator illuminates steady for 20 seconds before it flashes, activates MASTER CAUTION lights.
	**L/R AC BEARING** Annunciator illuminates steady to indicate the respective AC alternator bearing failure impending within the next 20 hours of operation.		**STBY P/S HTR** ON GROUND — Annunciator illuminates steady to indicate the standby pitot-static heater switch is OFF. IN FLIGHT, annunciator flashes to indicate the stand-by pitot-static heater is off or inoperative, activates MASTER CAUTION lights.
	**RUDDER BIAS** Annunciator flashes to indicate rudder bias system malfunction, rudder bias system valve not in commanded position. Activates MASTER CAUTION lights.		**AOA HTR FAIL** ON GROUND, annunciator illuminates steady to indicate the AOA heater is OFF. IN FLIGHT, annunciator flashes to indicate the AOA heater is OFF or failed, activates MASTER CAUTION lights.
	**FIRE EXTINGUISHER BOTTLE LOW** Annunciator flashes if one or both engine fire extinguisher bottles have low pressure. Activates MASTER CAUTION lights.		**AIR DUCT O'HEAT** Annunciator flashes if air in the cockpit duct and/or cabin duct has exceeded temperature limits, activates MASTER CAUTION lights.
	**L/R FUEL FLTR** Annunciator flashes if the respective engine fuel filter bypass switch has activated to indicate an impending bypass condition due to possible filter blockage. Activates MASTER CAUTION lights.		**RADOME FAN** Annunciator flashes if the radome fan is not operating. Ground operating time is limited to 30 minutes, and not allowed to dispatch in IMC conditions. Activates MASTER CAUTION lights. Advisory — Indicates the optional Flight Data Recorder is inoperative.
	**LO BRK PRESS** ON GROUND, annunciator flashes if hydraulic brake pressure is low, activates MASTER CAUTION lights. IN FLIGHT, annunciator illuminates steady for 20 seconds before it flashes, activates MASTER CAUTION lights.		**L/R TAIL DEICE FAIL** Annunciator flashes after the system is selected on, and one of the following failures occurs: • Tail deice valve has a loss of voltage. • Tail deice system has a loss of pressure during a six second cycle ON time. • Activates MASTER CAUTION lights.

Table 4-1 ANNUNCIATOR ILLUMINATION CAUSES (cont.)

ANNUNCATOR	CAUSE FOR ILLUMINATION	ANNUNCATOR	CAUSE FOR ILLUMINATION
TAIL DEICE PRESS L | R	**L/R TL DEICE PRESS** Annunciator illuminates if the respective tail deice is operating properly.	W/S O' HEAT L | R	**L/R W/S O'HEAT** Annunciator flashes if an overheat condition is detected in the respective windshield, activates MASTER CAUTION lights.
------ FUEL XFEED	**FUEL XFEED** Annunciator illuminates steady if fuel crossfeed is selected and the fuel crossfeed valve is open. Annunciator flashes and master caution illuminates steady if fuel crossfeed is selected off and the fuel crossfeed valve is not closed.	F / W SHUT OFF L R	**L/R FW SHUTOFF** Annunciator flashes if the respective engine hydraulic and fuel firewall shut off valves have both closed, activates MASTER CAUTION lights.
FUEL BOOST L | R	**L/R FUEL BOOST** IN FLIGHT — Annunciator illuminates steady for 10 seconds and then begins flashing if the boost pump is energized automatically due to low fuel pressure, and the throttle is out of cutoff. Activates MASTER CAUTION lights. If boost pump(s) energized for other than low fuel pressure, the annunciator illuminates steady with no illumination of MASTER CAUTION lights.	FIRE DET SYS L | R	**L/R FIRE DET SYS** Annunciator flashes if the engine fire detect system fails, activates MASTER CAUTION lights.
LO FUEL PRESS L | R	**L/R LO FUEL PRESS** Annunciator illuminates steady if the fuel system has low pressure prior to engine starts while the aircraft is on the ground. Annunciator flashes and master caution illuminates steady, if the fuel system has low pressure after both engines are started with aircraft on the ground or in flight and the throttle is out of cutoff. Activates MASTER CAUTION lights.	ACC DOOR UNLOCK NOSE | TAIL	**NOSE/TAIL ACC DOOR UNLOCKED** NOSE/TAIL annunciator flashes if the respective L and/or R nose door or tailcone and/or baggage door is not latched properly, activates MASTER CAUTION lights.
W/S FAULT L R	**L/R W/S FAULT** Annunciator illuminates steady eight seconds then will begin flashing if the windshield temperature controller has detected a fault prior to both engine starts while the aircraft is on the ground. Activates MASTER CAUTION lights. Annunciator flashes if the windshield temp controller detects a fault after engine starts with aircraft on the ground or in flight. Activates MASTER CAUTION lights.	DOOR SEAL	**DOOR SEAL** Annunciator steady on ground, flashes in flight, if the door seal pressure drops below 5.5 psi, activates MASTER CAUTION lights. Annunciator will extinguish if door seal pressure increases to approximately 8.5 psi.
		CABIN DOOR	**CABIN DOOR** Annunciator flashes to indicate the cabin door is not properly locked (possible disengagement of a cabin door pin). Check visual indicators on door frame. Annunciator remains steady on ground prior to engine start. Annunciator flashes after engine starts and in flight, activates the MASTER CAUTION lights.

Table 4-1 ANNUNCIATOR ILLUMINATION CAUSES (cont.)

ANNUNCATOR	CAUSE FOR ILLUMINATION	ANNUNCATOR	CAUSE FOR ILLUMINATION
	**EMER EXIT** Annunciator flashes if the emergency exit door is not properly secured and locked. Activates MASTER CAUTION lights.		**L/R WING ANTI-ICE** ON GROUND, annunciator illuminates steady when the wing/engine anti-ice is initially selected on and extinguishes when normal operating temperature is reached. After the light extinguishes, an undertemp condition will cause annunciator to flash and activate MASTER CAUTION lights. IN FLIGHT, annunciator initially illuminates steady until operating temperature is reached. The light flashes if the respective wing bleed air temp does not reach operating temperature within 4 minutes and 45 seconds after initially selecting ON. After the light extinguishes, an under temp condition will immediately cause the annunciator to flash and activate the MASTER CAUTION lights.
	**LAV DOOR** Annunciator flashes if the lavatory door(s) is/are not latched open prior to takeoff, or in flight with the flaps not up (zero degrees). Activates MASTER CAUTION lights.		
	**L/R BLD AIR O'HEAT** Annunciator flashes if the bleed air precooler (engine pylon mounted) output temperature has exceeded 560°F, activates MASTER CAUTION lights. If the wing A/I is ON, the respective wing A/I, L or R, will automatically shutdown.		
	**CHECK PFD 1, CHECK PFD 2** Annunciator flashes to indicate a Display Unit Wrap Around failure between the #1 IAC and pilot's PFD and/or the #2 IAC and Copilot's PFD. Activates MASTER CAUTION lights.		
	**L/R WING O'HEAT** Annunciator flashes if the inboard wing spar temperature (behind heat shield) exceeds limits (>160°F). Normally, indicates a bleed-air leak in the heat shield. Activates MASTER CAUTION lights.		

AUDIO WARNING SYSTEM

Various audio warnings are incorporated into the airplane systems that warn of specific conditions and malfunctions.

Testing the audio system and various other warning systems functions is provided by the rotary test switch (Figure 4-1).

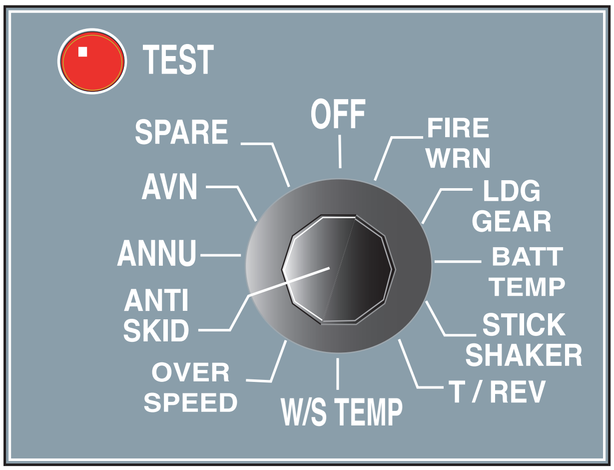


Figure 4-1 Rotary Test Switch

When the switch is rotated through each position, the associated system audio and annunciator test functions is described in Table 4-2.

TEST SYSTEM

The test selector is located on the center pedestal directly below the throttle quadrant and offers several positions of test (Figure 4-1). It will function only when the BATT switch is ON. A red light above the test selector switch illuminates whenever the test selector switch is in any position but OFF.

Table 4-2 TEST INDICATIONS (cont.)

SWITCH POSITION	INDICATION
OFF	• The red light will be extinguished and the test system is OFF. NOTE: The red light above the rotary test switch should illuminate for all the other test positions, including the spare position.
FIRE WARN	• LH ENG FIRE and RH ENG FIRE lights should illuminate.
LDG GEAR	• The green LH, RH and NOSE lights should illuminate. • The red GEAR UNLOCK light should illuminate. • The gear warning horn should sound.
BATT TEMP	• The red BATT O'TEMP->160° annunciator should "flash." • The battery temperature gage should indicate 160°F. • The MASTER WARNING lights should illuminate (cancelable).
STICK SHAKER	• Stick shaker should activate immediately on both control columns. • The AOA gage needle should swing to the top of the red band. • The red chevron in the AOA indexer should "flash."
THRUST REVERSER	• The ARM, UNLOCK and DEPLOY thrust reverser indicators lights should illuminate "steady." • The MASTER WARNING lights should illuminate (cancelable).
W/S TEMP	• With windshield heat selected on, the W/S O'HT annunciator should illuminate 3 to 4 seconds, then extinguish. NOTE: Test with the engines shut down, the W/S FAULT annunciator should illuminate and remain on (AC alternators are not supplying power).
OVER SPEED	NOTE: The avionics switch must be on for check of overspeed warning horn and related EFIS display information. • The MADC output reverts to Functional Test Mode and PFD1 and PFD2 should indicate: • 265 KIAS • Mach 0.4 • 5,000 feet altitude • And a vertical speed of 2,000 FPM climb • The audible overspeed warning should sound.
ANTISKID	• With the antiskid switch on, the ANTISKID INOP annunciator should flash for six seconds, then extinguish. • The MASER CAUTION light should illuminate during the self-test.
ANNUNCIATOR	NOTE: The avionics switch must be on for the annunciator check. • All lights on the annunciator panel should illuminate. • MASTER WARNING and MASTER CAUTION lights should illuminate (not cancelable). • Both red turbine overspeed lights should "flash." • Engine instrument LCDs should show "steady" 8s. • The AP OFF and YD OFF annunciators should illuminate. • The Flight Director Mode Selector (FDMS) buttons should illuminate left to right and then remain "steady." • The annunciators to the right of the F/D mode selector panel should illuminate "steady," (Figure 4-2). The lights are listed below, but may vary depending on installed options: 1. FD/AP PFD 1, FD/AP PFD 2 2. TERR NORM, TERR INHIB

Table 4-2 TEST INDICATIONS (cont.)

SWITCH POSITION	INDICATION
ANNUNCIATOR (cont)	3. GPWS FLAP NORM, GPWS FLAP O'RIDE 4. GPWS G/S, O'RIDE 5. GPWS TEST 6. AUDIO SPK/HPH 7. PHONE CALL • All A/P control panel lights should illuminate steady. • The green A/C (air conditioner) ON light above the A/C switch should illuminate "steady." • The Altitude Alert aural warning tone should sound.
AVIONICS	NOTE: The avionics switch must be ON for the avionics check. • MASTER CAUTION should illuminate (cancelable). • The Flight Director Mode Selector (FDMS) buttons should illuminate left to right and then remain on steady. • All A/P control panel lights should illuminate steady. • After a short delay, the lights listed below on the annunciator panel should "flash," indicating a successful self-test. 1. AP PITCH MISTRIM 2. AP ROLL MISTRIM 3. RADOME FAN 4. CHECK PFD 1 5. CHECK PFD 2 • The annunciators to the right of the F/D mode selector panel should illuminate "steady," (Figure 4-2). The lights are listed below, but may vary depending on installed options: 1. FD/AP PFD 1, FD/AP PFD 2 2. TERR NORM, TERR INHIB 3. GPWS FLAP NORM, GPWS FLAP O'RIDE 4. GPWS G/S, O'RIDE 5. GPWS TEST 6. AUDIO SPK/HPH 7. PHONE CALL • The AP OFF and YD OFF annunciators should annunciate "steady." • The autopilot disconnect warning aural horn should sound ON and OFF continually until the switch is moved. NOTE: If an optional phone is installed and a PHONE CALL switchlight is installed in the strip of annunciators, a "pulsating" phone call tone will sound during the test. Depressing the PHONE CALL switchlight will cause the tone to stop pulsating and become "steady."
SPARE	• This is an unused position and should not activate any system.

Figure 4-2 Avionics Switch Lights

CHAPTER 5
FUEL SYSTEM

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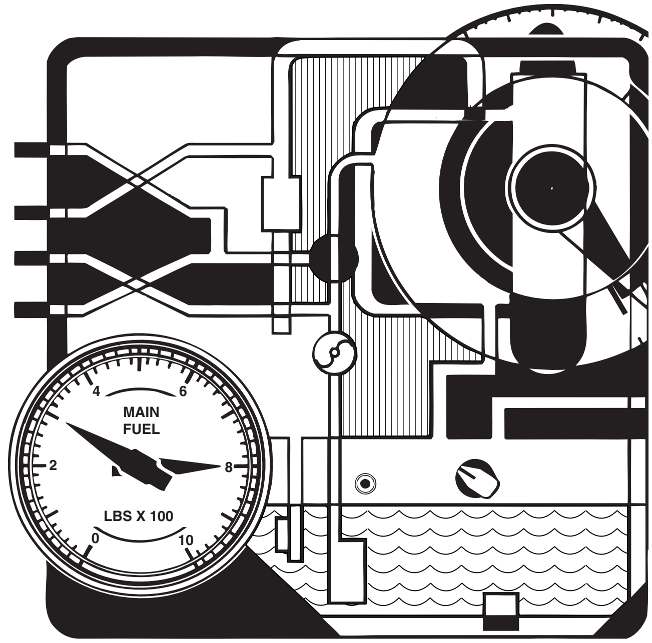
CHAPTER 5
FUEL SYSTEM



INTRODUCTION

This chapter details the Citation Excel fuel system. The wing contains two separate fuel tanks, left wing and right wing. Each tank normally supplies its respective engine; however, a fuel crossfeed system is incorporated.

GENERAL

Fuel flow to the engines from each wing tank is normally accomplished by motive flow charged primary ejector pumps or by electric boost pumps. The system is controlled by two boost pump switches and a XFEED selector knob on the pilot's instrument panel, and monitored by annunciator lights and quantity indicators. For description and operation of the engine fuel system, refer to Chapter 7, POWERPLANT.

DESCRIPTION AND OPERATION

FUEL STORAGE

The wing structure contains two integral sealed fuel tanks. Each wing tank has a usable fuel capacity of approximately 503 gallons or 3,395 lbs. Both tanks full equal a total capacity of 1,006 gallons or 6,790 lbs. The tanks are interconnected by a crossfeed line which is

opened by a crossfeed valve in the left wing tank, center section.

Each tank includes all wing area between the forward and rear wing spars, except over the main landing gear wheel wells and the extreme wing tip area (Figure 5-1). Lightening holes and stringer cutouts permit fuel movement within the wings. Flapper-type check valves are installed in the outboard wing rib assemblies to prevent a rapid shift of fuel to the outboard section of the wing when the airplane is in a wing low attitude, but allows free flow inboard.

A sealed engine feed hopper is located in the inboard sump section of each wing. The hopper is vented at the top to maintain a full hopper under low fuel conditions. It has flapper-type check valves that allow for grav-

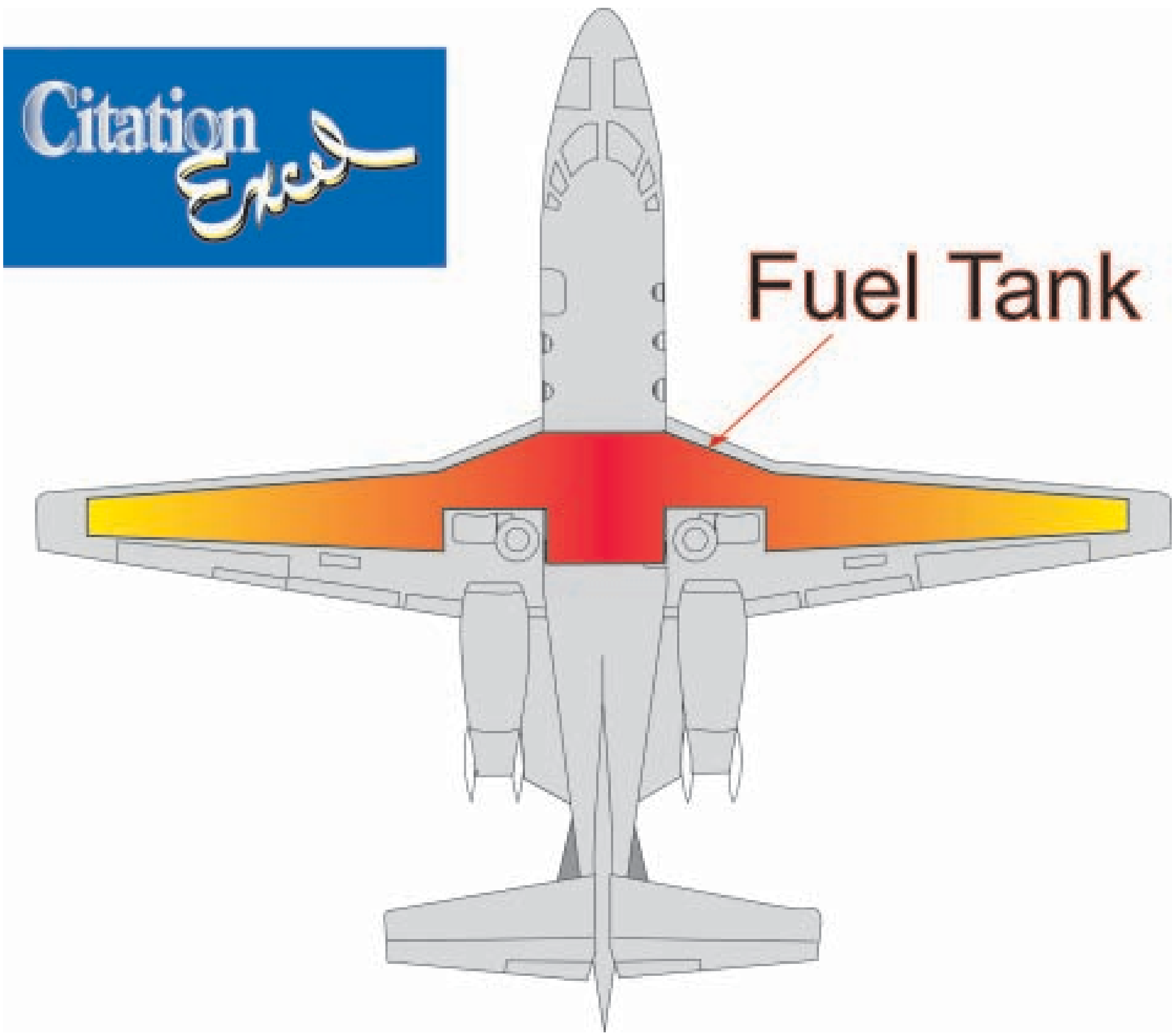


Figure 5-1   Excel Fuel Tanks Location

ity fuel flow into the hopper. Components that directly feed fuel to the engine are located in the hopper, i.e., primary ejector pump and electric boost pump.

In addition to the hopper and its components, each tank includes scavenge ejector pumps, a vent system, drain valves, relief valve (pressure and vacuum), over wing filler cap, single-point refueling/defueling components, and fuel quantity capacitance probes.

Tank Vents

A ventilation system is installed in each wing to maintain positive internal tank pressures within the structural limits of the wing. A surge tank, vent float valve and a vent scoop assembly are located near the wing tip areas. A vent line extends from the surge tank to the sump area in the center wing sections. The inboard end of the line is open and provides an entry for air if check valves and float valves fail in the closed position.

The surge tank is semi-isolated from the remainder of the wing fuel tank, and does not normally contain fuel. The surge tank functions as a fuel collector for small amounts of fuel which may be trapped in the climb vent line during flight maneuvers and climb attitudes or thermal fuel expansion.

The surge tank is vented to atmosphere by the vent scoop located on the lower wing surface (Figure 5-2). The vent scoop is connected to the surge tank with an open-ended tube located at the high point in the surge tank preventing fuel from siphoning overboard. It also prevents fuel from spilling overboard in flight during wing low conditions or uncoordinated turns. Check valves located on the surge tank allow fuel accumulations to drain back to the wing tank.

Figure 5-2 Fuel NACA Vent Scoop

The vent float valve is installed on the upper section of the wing structure between the main tank and the surge tank which allows air to either enter or leave the main fuel cell via the surge tank. It acts as the primary vent for level attitudes, including refueling and defueling. The valve is float actuated such that whenever fuel moves to the wing tip for any reason, the valve closes preventing fuel flow into the surge tank. If the airplane is parked on a sloping ramp, the vent float valve on the downward wing may close and fuel expansion may force fuel through the open end of the vent tube and out the vent scoop, thus preventing excessive pressure buildup in the low wing.

Relief Valve

A positive/negative pressure relief valve is installed on the underside of each wing (Figure 5-3). This valve protects the fuel tanks from excessive pressure either positive or negative during refueling or defueling operations and as a vent backup in case of vent system failure.

Fuel Tank Filler

One flush-mounted fuel filler cap and adapter is located on the upper surface of each wing near the wing tip (Figure 5-4). The filler cap

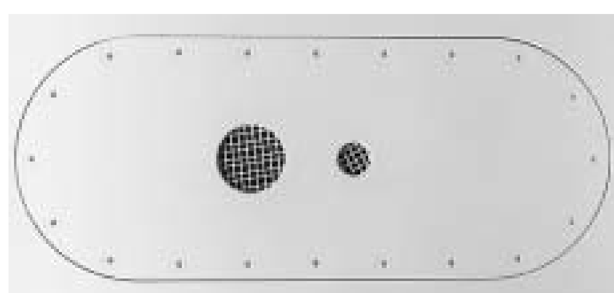


Figure 5-3 Pressure Relief Valve

CLOSED

OPEN

Figure 5-4 Fuel Filler Cap

and adapter consists of a key locking cap, adapter, and a safety chain that attaches the cap to the adapter. Each cap is recessed and marked to indicate "open" and "closed."

To open, unlock the cap and rotate the cap counterclockwise. To close, reverse the procedure. The locking caps are keyed-alike pairs. The keys are identified with the word **FUEL** and can be removed from the unlocked cap. A bright chrome-plated cap cover is installed over the cap to protect the lock from inclement weather.

During over-the-wing fueling, fuel level is controlled by the location of the fuel filler cap and fuel filler standpipe. Over fueling the wing tank will allow fuel to fill the standpipe and cause fuel to flow out the fuel filler cap opening. This assures the standpipe expansion space cannot be filled with fuel.

NOTE

During over-the-wing fueling, top off wing fueling only to the bottom level of the standpipe to prevent spillage.

Drain Valves

Fuel tank drain valves are located in the lower surface of each wing, four mounted in line, fore and aft next to the center wing skid and one located outboard of the wheel well (Figure 5-5) . The valves are tool operated, poppet type, and semi-flush external mounted. The valves allow draining of sediment, moisture, and/or residual fuel from the tanks.

MAJOR COMPONENTS

Electric Boost Pumps

One 28-VDC boost pump in the engine feed hopper of each fuel tank will supply fuel to the respective engine-driven fuel pump in concert with the primary ejector pump or as a stand alone pump during emergency/abnormal situations. The pumps are controlled by switches on the pilot's switch panel (Figure 5-6). The pumps are powered by main DC electric through circuit breakers located on the

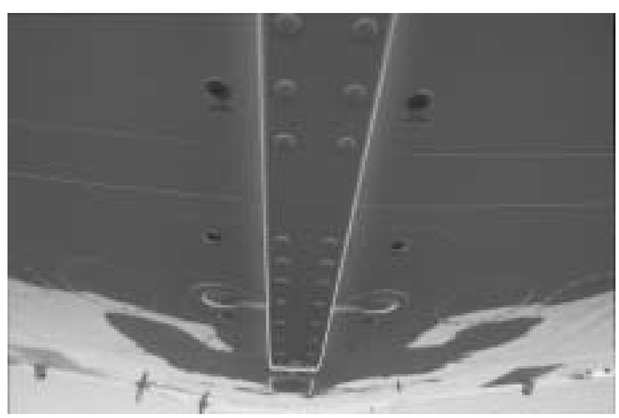


Figure 5-5 Fuel Drain Valves

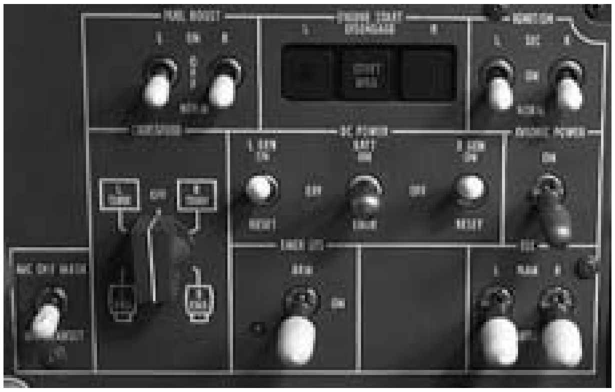


Figure 5-6 Pilot's Switch Panel

pilot's CB panel. Pump operation in each switch position is as follows:

ON — Boost pump operates continuously. **L** or **R FUEL BOOST** illuminates "steady."

OFF — Boost pump is deenergized. The boost pump will not operate regardless of other sensors.

NORM — Boost pump automatically activates in the following circumstances:

During engine start as the respective start button is depressed, the **L** or **R FUEL BOOST** annunciator illuminates "steady" and extinguishes at starter cutout speed, approximately 40-45% N_2.

During crossfeed, the boost pump operates in the tank selected. **L** or **R FUEL BOOST** illuminates "steady."

When low pressure is detected by the low fuel pressure switch, approximately 5 psi (**L** or **R LO FUEL PRESS** illuminates), and the throttle is out of the cutoff position, the boost pump will activate (**L** or **R FUEL BOOST** illuminates "flashing").

If the boost pump is supplying sufficient pressure, the **L** or **R LO FUEL PRESS** annunciator will extinguish. Operation continues until the switch is cycled to OFF and back to NORM, or retarding the throttle to cutoff.

NOTE

The electric boost pumps are designed so that its cartridge element for the motor and impeller can be replaced without tank entry or defueling.

Primary Ejector Pumps

One primary ejector pump is located in the engine feed hopper of each wing tank. The pumps supply the engine-driven fuel pumps with a continuous supply of low pressure high volume fuel at the required pressure and flow rate. They, in turn, are charged by high-pressure low-volume motive flow fuel, 425 to 725 psig, from the engine-driven pumps.

The primary ejector pumps also provide low-pressure motive flow for the three scavenge ejector pumps in each wing tank.

Scavenge Ejector Pumps

Each wing tank has three scavenge ejector pumps located in the sump area. They are continuous-operating ejector pumps that utilize motive flow from the primary ejector's discharge flow. The locations of the scavenge ejector pumps provides a continuous flow of fuel to the engine feed hopper, keeping it full in all normal flight attitudes.

The scavenge ejector pump inlets and feed hopper gravity inlets are protected by large area screens of wire mesh which minimize contamination reaching the hopper and fuel system components.

Crossfeed Valve

The crossfeed valve is a motor-operated ball valve, electrically driven open and closed during crossfeed operations. The valve is installed in the plumbing connecting the left and right engine feed manifolds. The crossfeed valve is located in the left feed hopper and powered by the main DC electrical system through the FUEL CONTROL circuit breaker on the pilot's CB panel.

Motive Flow Shutoff Valves

Motive flow shutoff valves are installed in each motive flow line to shut off primary ejector pump motive flow to the non-feeding fuel tank when crossfeed is selected. The valves are normally open, electrically closed solenoid valves.

Firewall Shutoff Valves

Engine fuel firewall shutoff valves are installed in the wing fairing areas in each fuel supply line. The valves are electric motor-operated ball valve assemblies that shut off fuel flow to the engine(s) in event of an engine fire. They are activated closed or open by pressing the corresponding guarded red left or right engine fire switch (LH ENGINE FIRE and RH ENGINE FIRE) located below the glareshield on the firetray, to the left and right of the annunciator panel.

When the valve is closed, provided the hydraulic firewall shutoff valve is also closed, the corresponding **F/W SHUTOFF L** or **R** annunciator will illuminate "flashing."

NOTE

> If an engine is shut down in flight for reasons other than fire, the shutoff valve must be left open and the boost pump operated to prevent damage to the engine-driven fuel pump.

CONTROLS

Controls for the fuel system are located on the pilot's instrument panel (Figure 5-6). The L and R FUEL BOOST pump switches control the electric boost pumps. Detailed explanation of the boost pumps is presented under MAJOR COMPONENTS, Electric Boost Pumps, earlier in this chapter.

The CROSSFEED selector is located on the pilot's switch panel directly below the FUEL BOOST pump switches (Figure 5-6). The CROSSFEED selector has three positions labeled "L TANK to R ENG–OFF–R TANK to L ENG."

Moving the selector from OFF to either tank, selects the tank from which fuel is to be extracted from and the engine to be supplied. With both engines operating, the tank selected will still supply fuel to its corresponding engine and to the opposite engine. Any excess fuel flow will be sent to the opposite wing tank through the scavenge ejector pumps.

Detailed explanation of the fuel system during normal and crossfeed operation is presented under OPERATION later in this chapter.

INDICATING SYSTEM

Quantity Indication

The fuel quantity indicating system is a capacitance system consisting of a dual linear fuel quantity indicator, a fuel quantity signal conditioner with self-test and monitoring features, and seven fuel probes (sensing units) per wing (Figure 5-7). The fuel probes are located so that accurate indications of fuel volume is maintained during all flight attitudes. Each fuel probe has an integral electronic module that converts capacitance to a useable signal to the signal conditioner.

The Fuel Quantity Signal Conditioner (FQSC) is a dual channel microprocessor based unit that has a channel for the left wing and right wing fuel systems. It interfaces with all the wing fuel probes and the fuel quantity/fuel flow indicator. The FQSC is mounted in the cockpit behind the pilot's side console.

The fuel quantity/fuel flow indicator is located on the center instrument panel (Figure 5-8). It indicates actual useable fuel remaining in the left and right wing tanks, and the fuel flow rate of each engine.

The low fuel warning system consists of two low fuel level float switches and two indicator lights on the annunciator panel, LO FUEL LEVEL L/R. One low fuel level float switch is installed in each wing tank.

The float system consists of a float in each fuel tank and an electrical switch mounted in the

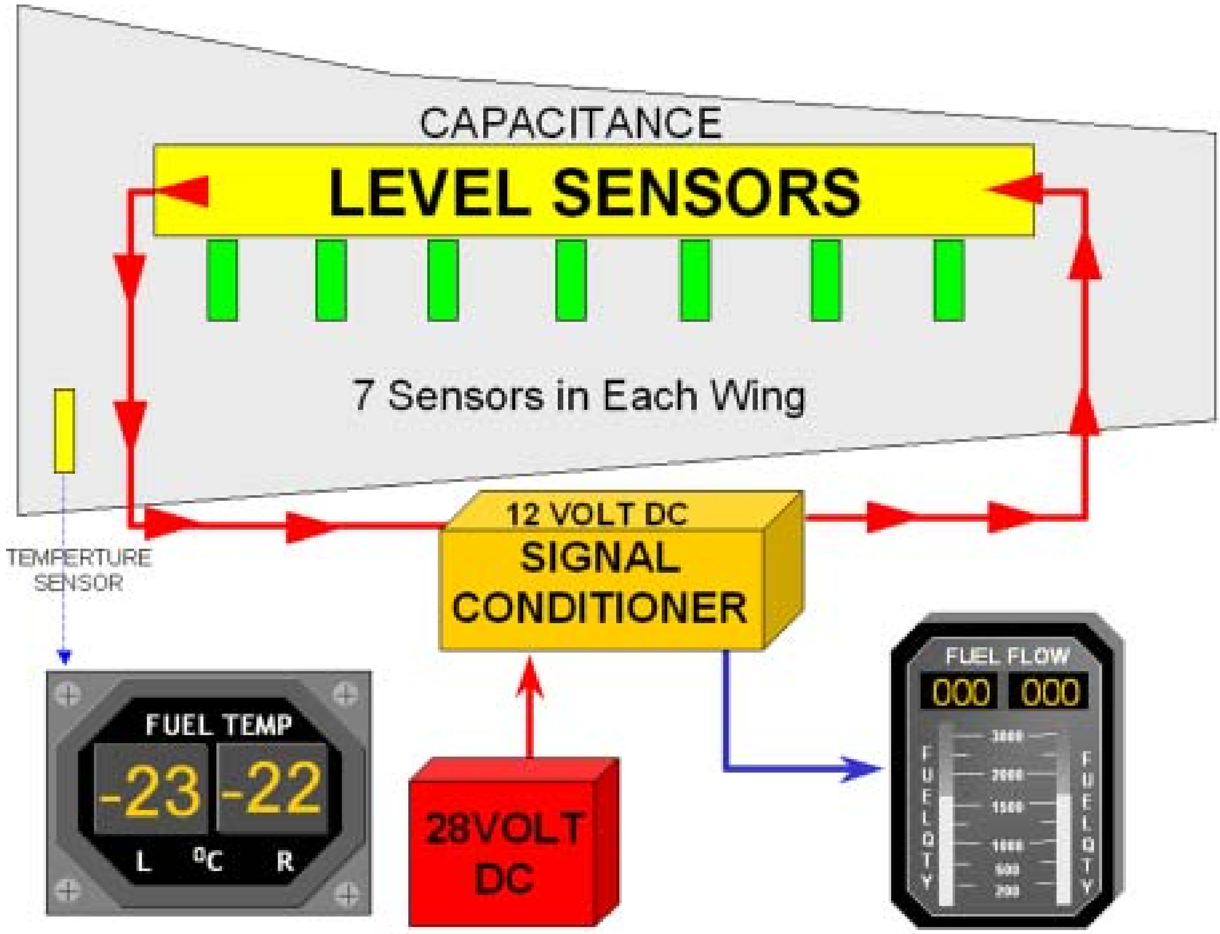


Figure 5-7 Fuel Gauging System

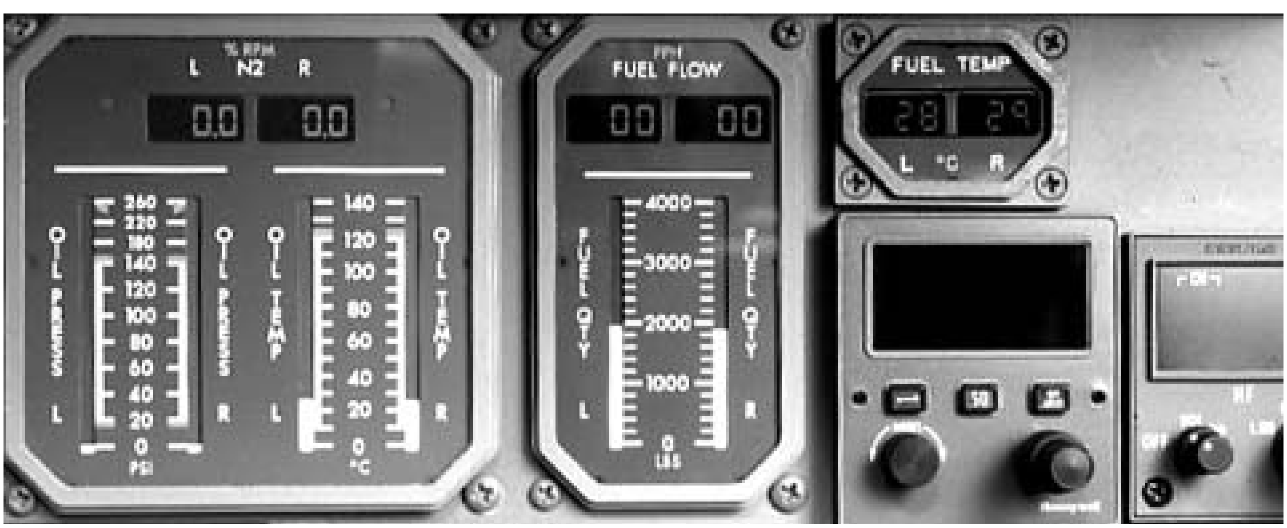


Figure 5-8 Fuel Quantity/Fuel Flow/Fuel Temp Indicators

wheel wells. When total fuel level in a wing tank decreases to approximately 360 ± 20 lbs, the float drops and actuates the electrical switch mechanically, and after 30 seconds, causes the respective **LO FUEL LEVEL L** or **R** annunciator to illuminate flashing.

Operation of the Fuel Quantity Indication System

The FQSC supplies an electrical signal to each fuel-sensing probe, they in turn send capacitance signals back to the FQSC. The FQSC processes the sensing probe signals electronically and provides two output signals for use by the L and R FUEL QTY indicators. The fuel quantity indicator convert the signals from the FQSC into a linear scale indication, U.S. pounds, fuel per wing (Figure 5-8).

A built-in test (BIT) function of the fuel quantity signal conditioner checks each fuel probe signal for validity. A failure is annunciated on the FQSC by use of three light-emitting diodes (LED). A detected failure will cause the L and/or **R FUEL GAUGE** annunciator to illuminate flashing.

The BIT test function also self-monitors the FQSC for circuit faults. If failures are detected, the channel's discrete BIT Fault Output will be turned ON, and the BIT status LEDs will display a pattern that identifies the failure. The LED display pattern remains on until electrical power is removed from the FQSC.

NOTE

If a L and/or **R FUEL GAUGE** annunciator illuminates, do not secure aircraft main DC electrical power until maintenance records the FQSC BIT indications.

The complete indicating system operates on main DC power through the L and R FUEL QTY circuit breakers on the pilot's CB panel.

Fuel Flow

FUEL FLOW Liquid Crystal Display (LCD) indicators are located above the FUEL QTY indications on the same indicator (Figure 5-8). The LCDs indicate fuel consumption in pounds per hour per engine. Fuel flow is transmitted to the indicator by fuel flow meters on the engines downstream from each engine fuel control unit (FCU).

Fuel Temperature Indication

Fuel temperature is measured by fuel temperature sensors installed in the sump area of each fuel tank. The temperature reading is sent to the fuel temperature indicator (FUEL TEMP L/R) located on the center instrument panel (Figure 5-8). Temperature is indicated by LCD display for each fuel tank and indicates temperature in Celsius with a range from -60°C to +70°C, with a tolerance of ±3° C.

Annunciator Lights

The following seven annunciator lights are associated with the fuel system (refer to Chapter 4, MASTER WARNING SYSTEM).

FUEL GAUGE L/R will illuminate flashing to indicate the fuel quantity signal conditioner (FQSC) detected a fault in the fuel quantity sensing system. If a fault is annunciated, ensure the FQSC BIT indicators are checked by maintenance personnel after engine shutdown but before main DC electrical power is secured.

F/W SHUTOFF L/R light illuminates flashing when both fuel and hydraulic firewall shutoff valves close when the respective LH or RH ENGINE FIRE switch-light is depressed. Depressing the respective ENGINE FIRE switch-light a second time will open the shutoff valves and the annunciator light will extinguish. Refer to Chapter 8, FIRE PROTECTION for more detailed information.

LO FUEL PRESS L/R annunciators illuminate when fuel pressure drops below approximately 5 psi and extinguishes as fuel pressure

increases above approximately 8 psi. The low fuel pressure switch will activate the respective electric boost pump if its switch is in the NORM position. As fuel pressure increases, the respective **LO FUEL PRESS L/R** light will extinguish.

On the ground prior to engine start, the **LO FUEL PRESS L/R** annunciators will remain steady (MASTER CAUTION lights extinguished). The electric boost bumps are disabled with the throttles in cutoff, unless boost pump switch(es) are ON.

LO FUEL LEVEL L/R annunciator(s) will illuminate flashing when useable fuel in the corresponding wing fuel tank drops to approximately 360 ± 20 lbs. The annunciator light is activated by a float switch in the wet area of the respective fuel tank. The float has to drop and actuate the electrical switch for 30 seconds to cause the **LO FUEL LEVEL L/R** light to illuminate.

FUEL BOOST L/R annunciator will normally illuminate steady anytime the boost pump is energized. In flight, the annunciator will illuminate flashing after 10 seconds of steady illumination if the boost pump is actuated by the low fuel pressure switch (below approximately 5 psi) and the respective throttle is out of the cutoff position.

FUEL FLTR BP L/R annunciator will illuminate flashing if a differential pressure drop across the engine-mounted fuel filter is sensed due to possible contamination. The light indicates the respective fuel filter has or is on the verge of bypassing fuel. The filter element must be inspected after landing.

FUEL XFEED advisory light illuminates steady to indicate fuel crossfeed has been selected and the crossfeed valve is open. When deselecting crossfeed and the crossfeed valve does not close, the **FUEL XFEED** advisory light will illuminate flashing and activate the MASTER CAUTION lights steady.

OPERATION

Normal

(Figure 5-9) illustrates the fuel system.

Engine Starting

With the FUEL BOOST pump switch in NORM, depressing an engine start button activates the respective electric boost pump in the engine feed hopper. It provides high-volume, low-pressure flow of fuel to the engine-driven fuel pump. The boost pump supplies head pressure to the engine-driven fuel pump, which in turn develops high-pressure fuel for the fuel nozzles. The nozzles atomize fuel required for combustion.

During the initial start process (depressing the start button), the corresponding **FUEL BOOST L/R** will illuminate steady and the associated **LO PRESS L/R** annunciator will extinguish.

When engine start terminates at approximately 40-45% N_2, the fuel boost pump will shut off and the respective **FUEL BOOST L/R** annunciator will extinguish.

Engine Operation

The engine-driven fuel pump supplies high pressure, low-volume, motive flow fuel through the motive flow line and the normally open motive flow shutoff valve to the respective primary ejector pump. Prior to engine start termination, the primary ejector pump is charged by the engine-driven fuel pump to ensure the engine will operate on its own after the electric boost pump deenergizes.

The primary ejector pump is the main source of continuous fuel supply to the engine-driven pump to provide head pressure to maintain engine operation.

The primary ejector pump also supplies high-volume, low-pressure, fuel to the three scav-

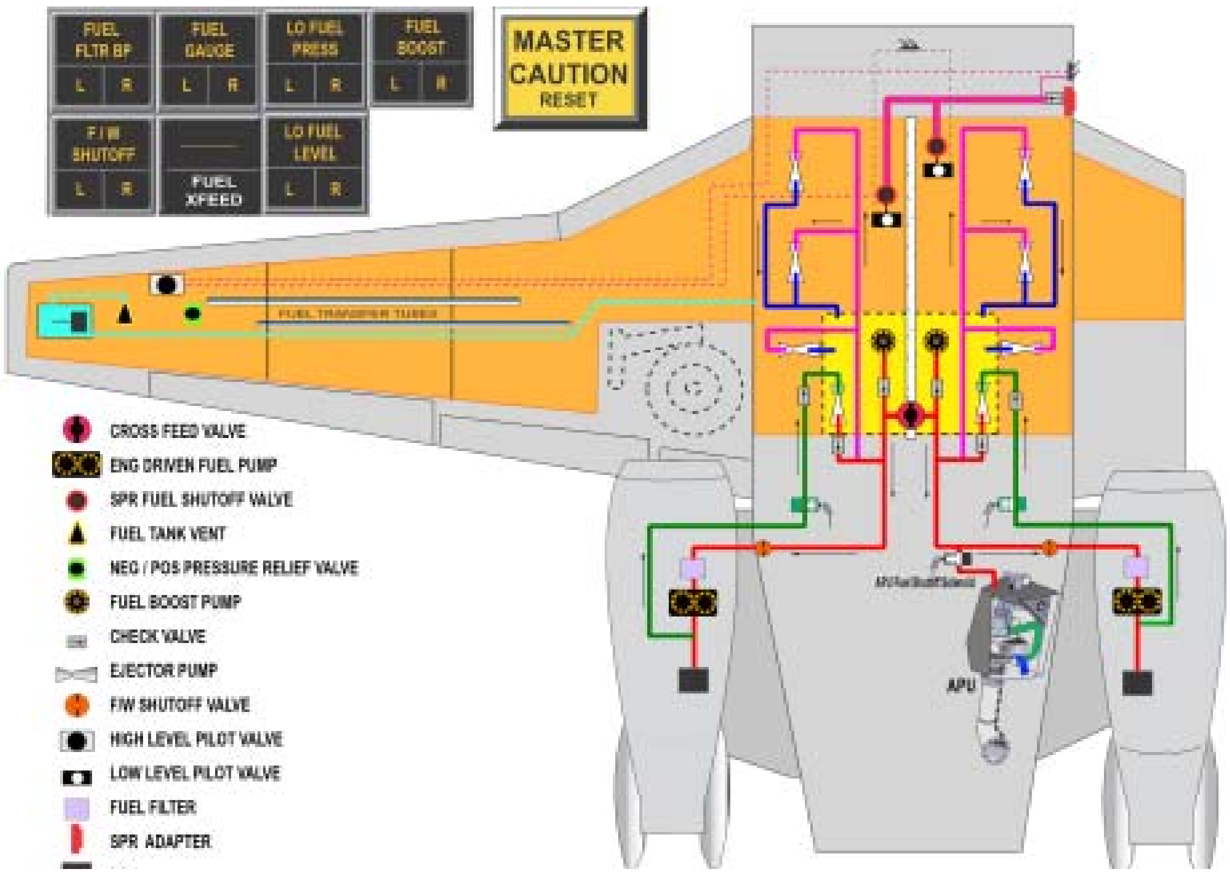


Figure 5-9 Fuel System Schematic

enge ejector pumps in the sump area to keep the engine feed hopper full.

Refer to Chapter 7, POWERPLANT, for a more detailed discussion of engine operation.

Crossfeed

Fuel crossfeed is normally required to correct a fuel imbalance between wing fuel tanks.

The engine/fuel crossfeed system allows either engine or both engines to be fed from the primary ejector and/or electric boost pump in either tank. Crossfeed components include a motor-driven crossfeed ball valve, motive flow solenoid operated shutoff valves, and associated plumbing.

Placing the CROSSFEED selector to either L TANK to R ENGINE or R TANK to L ENGINE causes the crossfeed valve to power open and activates the corresponding electric boost pump in the tank selected. The respective **FUEL BOOST L/R** annunciator will illuminate steady. When the crossfeed valve opens, the **FUEL XFEED** advisory annunciator will illuminate steady. Approximately three seconds after the crossfeed selector is positioned, the opposite side motive flow shutoff valve energizes closed.

Closure of the motive flow shutoff valve terminates normal fuel flow, and allows selected tank pressure to provide fuel flow to the opposite engine fuel manifold. This action also allows fuel from the selected tank to flow through the three scavenge pumps in the opposite tank thus transferring fuel to the opposite side wing tank.

NOTE

If the motive flow shutoff valve closed too soon, a low pressure spike could occur and activate the electric boost pump on the receiving side, restricting fuel flow from the sending tank to the receiving side engine and tank. Such an occurrence is indicated by illumination of both **FUEL BOOST L/R** annunciators. Crossfeed is not possible in this situation.

NOTE

If one boost pump switch is OFF and crossfeed is selected to that tank, the crossfeed system will not operate (crossfeed valve will not open and opposite motive flow valve will not close). Selecting the opposite tank with the boost pump switch NORM, the crossfeed valve will open, the opposite motive flow valve will close and the respective engine will shutdown (boost pump switch OFF). If both boost pumps are selected OFF, the crossfeed system will not operate regardless of which tank is selected.

NOTE

The fuel quantity gages must be monitored to ensure crossfeed is actually occurring.

The fuel transfer rate from tank-to-tank will vary depending on fuel requirements demanded by the engine(s). If fuel is transferred from tank-to-tank on the ground, (engines shutdown), flow rate is approximately 700-900 pounds/per/hour.

NOTE

If during refueling operations, a fuel imbalance occurs and crossfeed is selected to correct the imbalance, it is advisable to use a GPU or an APU to power the operation.

To terminate crossfeed operation, place the CROSSFEED selector to OFF. The motive flow shutoff valve immediately deenergizes open to establish normal operating fuel pressure on the receiving side, and three seconds later, the crossfeed valve closes. The respective **FUEL BOOST L/R** annunciator and the **FUEL XFEED** advisory light will extinguish and the fuel system will return to normal operation (corresponding fuel tanks to their respective engines).

NOTE

If after selecting crossfeed OFF, and if the crossfeed valve does not close, the **FUEL XFEED** advisory light will illuminate flashing and activate the MASTER CAUTION lights.

The crossfeed system is powered by main DC power through the FUEL CONTROL circuit breaker on the pilot's CB panel.

FUEL SERVICING

OVER WING FUELING

Over wing fueling is accomplished through flush filler caps installed on the top of each wing near the wing tip (refer to FUEL STORAGE, Fuel Tank Filler, earlier in this chapter).

Ensure fueling apparatus is properly grounded. Insert the fuel nozzle into the fuel standpipe and begin fueling. If a full wing tank is desired, cease fueling as the fuel reaches the bottom of the standpipe. Continuing fueling will fill the standpipe and result in fuel spillage on the wing. Ensure the fuel caps are properly secured (FWD stamped on the adapter facing toward the leading edge), locked and the cap positioned so the chrome cover can pushed down toward the rear.

SINGLE POINT PRESSURE REFUEL/DEFUEL SYSTEM

The Single Point Pressure Refueling (SPPR) system is used to fuel the wing tanks from a single refueling receptacle. Advantages of single point refueling are:

- Reduces refueling time.
- Reduces chances of fuel contamination.
- Protects the airplane from skin damage.
- Reduces static electricity hazards.
- Eliminates fuel contact with personnel.

The SPPR system is independent of over-the-wing fueling. Refueling is accomplished at the pressure refuel adapter (receptacle) located in the single point refuel/defuel compartment located in the fuselage forward of the right wing (Figure 5-10).

Description

The major components of the single point refuel system are the refuel/defuel compartment, pressure refuel adapter housing, refuel/defuel shutoff valves, and the precheck panel.

Refuel/Defuel Compartment

The SPPR compartment forward of the right wing contains the pressure refuel adapter and the precheck panel. The panel is equipped with a flush-mounted hinged access door. The door is hinged on the front and swings open forward. It is secured closed by two paddle latches on the rear of the door and a cam key lock. Ensure the door is closed, latched and locked (remove the key) prior to flight.

NOTE

The refueling/defueling door **is not** connected to a door unlock warning annunciator.

Refuel Adapter Housing

The Pressure Refuel Adapter Housing consists of an adapter and housing that allows refueling equipment to connect to the airplane. The adapter contains a spring loaded coupling valve that prevents fuel spillage during the coupling process. The housing has a port to supply precheck fuel flow to the precheck valves.

Refuel/Defuel Shutoff Valves

A Refuel/Defuel Shutoff Valve is installed in each wing tank. The valves shut off fuel flow during refueling when the tanks are full, and shut off fuel flow during defueling when the tanks are empty. The valves are spring loaded closed by positive pressure during refueling and pulled open by negative pressure during defueling. During refueling, part of the fuel flow is bypassed to a pilot line. When the tanks are full, the pilot port flow is cut off increasing back pressure to the shutoff valve causing it to close (Figure 5-9). During defu-

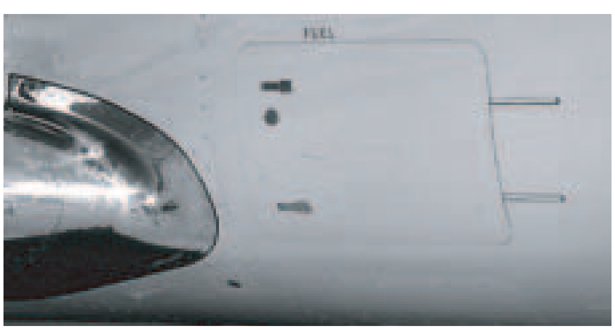


Figure 5-10 Single Point Refuel/Defuel Compartment

eling when a tank is near empty, the pilot port is opened to tank pressure and the respective shutoff valve closes.

Precheck Panel

The precheck panel is located in the Refuel/Defuel Compartment (Figure 5-10). The panel is equipped with two levers that allow service personnel to precheck fueling operations to ensure the system shuts down when wing tanks are full. Refueling and precheck procedures are listed on an instruction label inside the access door (Figure 5-11).

Figure 5-11 Refueling/Defueling Instructions (Door)

Refueling Operation

Single Point Pressure Refueling is accomplished by connecting the refueling equipment to the pressure refuel adapter (receptacle) in the refuel/defuel compartment (Figure 5-12). Fuel can be delivered to both wings simultaneously or to each wing independently.

During refueling, fuel is directed through a common manifold to each wing tank refuel/defuel shutoff valve. Fuel pressure opens the spring-loaded shutoff valves, delivering the majority of the fuel to the tanks.

Figure 5-12 Precheck Levers (Precheck Position)

A small quantity of fuel is bypassed to the high level pilot valve. As the fuel level reaches the high level pilot valve, a float-operated needle valve seats to close off pilot valve flow, building pressure on the back side of the shutoff valve. The resulting force imbalance closes the shutoff valve and fuel flow is terminated. If one wing fuel tank is filled before the other tank is full, flow shuts off to the full tank and the opposite wing tank will continue fueling until it is full.

Precheck Operation

Commence fuel flow and allow to stabilize. Pull out the precheck lever(s) and note the fuel flow should cease within 30 seconds (Figure 5-12). This check insures the shutoff valves will close when the tanks are full. *If the flow doesn't stop after 30 seconds have elapsed, secure SPPR operations and fuel over the wing.*

Opening either the left or right precheck valve will direct fuel to the precheck port of that wing's high level pilot valve. Fuel fills the float bowl faster than it can flow out, regardless of the fuel level in the tank. When the high-level pilot valve float becomes buoyant, the float-operated needle valve seats to close off the pilot flow in the wing tank. Fuel pressure in the pilot line closes and causes the refuel/defuel shutoff valve(s) to close.

Closing the precheck levers (valves) allows fueling to continue.

To prevent fueling a wing tank, raise the precheck lever on the wing tank not desired to be fueled.

CAUTION

IF REFUEL FLOW DOES NOT STOP DURING THE PRECHECK, REFUELING MUST BE IMMEDIATELY TERMINATED.

CAUTION

PRESSURE LIMITS ARE SHOWN ON A PLACARD AT THE SINGLE POINT PRESSURE REFUEL ADAPTER (RECEPTACLE).

CAUTION

MINIMIZE DURATION OF WING PRECHECK OPERATION WHEN THE WING TANKS ARE FULL; EXTENDED PRECHECK FLOW COULD CAUSE TANK(S) TO OVERFLOW.

Single-Point Defueling

Single-point defueling is accomplished by connecting the refueling equipment to the pressure refuel adapter. Application of negative pressure causes the refuel/defuel shutoff valves to open. Fuel is drawn from the tanks through the shutoff valves into a storage tank. Defueling is terminated when the fuel level lowers to the point where the low level pilot valve(s) floats drop, opening the pilot port to tank pressure which causes the refuel/defuel shutoff valves to close (Figure 5-9).

CAUTION

DEFUELING REQUIRES EQUIPMENT WITH ADEQUATE SUCTION AND HOSE STIFFNESS.

Defuel Select Valves

Manual defuel select levers are located on the front wing spar. The levers are accessed through a panel on the belly of the aircraft (Figures 5-13). During defueling operations, if both wing tanks are required to be defueled simultaneously, it is not necessary to open the access door and move the levers out.

CLOSED

OPEN

Figure 5-13 Manual Defuel Select Levers Access Door

NOTE

Moving either lever "out" will deactivate the corresponding refuel/defuel shutoff valve and prevent defueling the respective wing tank (Figures 5-14).

NORMAL OPERATION

PRELIMINARY COCKPIT INSPECTION

Check both FUEL BOOST pumps, NORM. The fuel boost pumps are normally left in NORM for the remainder of the mission. Check CROSSFEED switch OFF.

Ensure fuel quantity is sufficient for the projected flight.

NOTE

Maximum lateral fuel imbalance is 400 pounds. If imbalance exceeds 400 pounds, correct prior to flight.

EXTERIOR INSPECTION

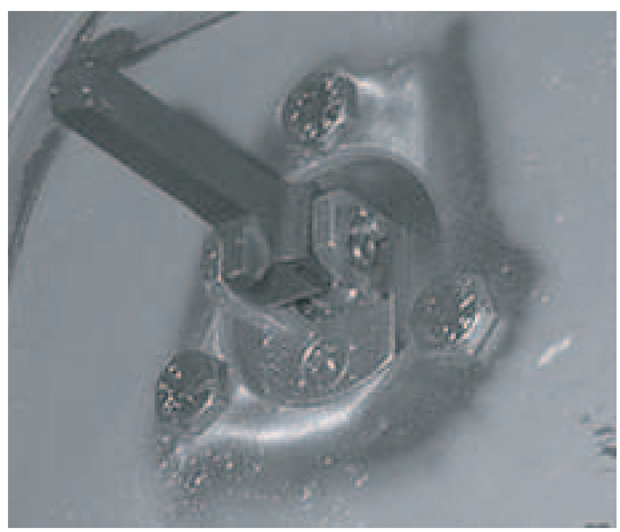
While checking the right side of the fuselage in front of the wing, check the SPPR door to ensure all latches are secured and it is key locked.

During each wing inspection, drain samples of fuel from the five quick drains and check for contamination. Push straight up on the drains. The drains may lock open if they are turned. Check the fuel filler caps locked down and secured properly (locking tab and/or cover pushed down toward the rear). Fuel tank NACA vents on the underside of the wings are clear.

IN FLIGHT

Monitor fuel flow indicators and fuel quantity gauges as appropriate.

ALLOW DEFUEL



PREVENT DEFUEL

Figure 5-14 Manual Defuel Select Lever

ABNORMAL OPERATION

LOW FUEL PRESSURE (LO FUEL PRESS L OR R CAUTION LIGHT ON)

If a **LO FUEL PRESS L** or **R** annunciator illuminates (**FUEL BOOST L/R** annunciator extinguished), turn the respective FUEL BOOST switch ON. If light doesn't extinguish, check the L and R FUEL BOOST circuit breakers IN on the pilot's CB panel.

Check for a possible fuel leak by observing the fuel quantity gages. Consider the possibility of initiating crossfeed to reestablish fuel pressure on the affected side.

LOW FUEL QUANTITY (LO FUEL L OR R CAUTION LIGHT ON)

Indicates the respective tank(s) fuel quantity is approximately 360 pounds remaining.

Turn the FUEL BOOST switch ON to ensure sufficient pressure is available for engine operation and for the scavenge pumps.

Check the L and R FUEL BOOST and FUEL CONTROL circuit breakers are IN on the pilot's CB panel.

Activate the crossfeed system as required.

Land as soon as possible.

FUEL BOOST PUMP ON (FUEL BOOST L OR R CAUTION LIGHT ON)

Indicates the respective fuel boost pump was either automatically or manually turned ON.

Turn the affected FUEL BOOST switch, ON then NORM. Check the respective **FUEL BOOST** annunciator segment to illuminate and extinguish (boost pump may have activated due to a momentary low fuel pressure spike).

NOTE

If affected FUEL BOOST caution light does not extinguish, refer to Abnormal Procedure, LOW FUEL PRESSURE. If operating with Jet B or JP-4 fuel, verify operations are within the Jet B/JP-4 Fuel Operating Envelope. Refer to LIMITATIONS Section of this manual.

FUEL FILTER BYPASS (FUEL FLTR BP L OR R CAUTION LIGHT ON)

Consider the possibility of partial or total loss of both engines thrust due to fuel contamination. Land as soon as practical.

Review **MAXIMUM GLIDE–EMERGENCY LANDING** Emergency Procedures in Section III of the *AFM* and/or the Abbreviated Checklist.

FUEL CROSSFEED (FUEL XFEED ADVISORY LIGHT ON)

Indicates the fuel crossfeed valve is open. The selected tank **FUEL BOOST L** or **R** annunciator should be illuminated if the system was selected and operating normal.

If the **FUEL XFEED** advisory light is "flashing" and the MASTER CAUTION lights are illuminated, indicates the crossfeed valve is open and the CROSSFEED switch is OFF.

FUEL GAGING SYSTEM FAULT (FUEL GAUGE L OR R CAUTION LIGHT ON)

Indicates that a fault has been detected in the respective fuel gaging system. Monitor the respective fuel gage for proper indication. Consider the possibility that the tank contains less fuel than the opposite tank. This fault may also be the result of improper fuel capacitance. Check fuel quantities after landing.

NOTE

After landing, the BIT. status LEDs on the fuel quantity signal conditioner, located in the LH wall of the cockpit, should be checked by appropriate personnel prior to battery switch OFF. Removal of power will reset the failure pattern displayed by the BIT status LEDs. Record fuel quantity in each tank at time the fault was initially displayed to assist in maintenance troubleshooting.

The FMS system can provide additional fuel quantity information provided it is programmed with accurate ramp fuel before engine starts.

LIMITATIONS

The fuels listed below are approved for use in accordance with Figure 2-5, Section II of the *AFM*:

- COMMERCIAL KEROSENE JET A
- JET A-1
- JET B
- JP-4
- JP-5
- JP-8 per CPW 204 specification

FUEL BOOST Pumps–ON; when low fuel level lights illuminate or at 400 lbs or less indicated fuel.

Fuel remaining in the fuel tanks when the fuel quantity indicator reads zero is not usable in flight.

SINGLE-POINT REFUELING

Single-point refueling must be accomplished as per the procedures contained in the placard located on the SPPR access door. Refueling pressure range is 10 to 55 psi, maximum defueling pressure is -10 psi.

TABLE 5-1 FUEL LIMITATIONS

	JET A, A-1, JP-5 & JP-8	JET B & JP-4
MINIMUM FUEL TEMPERATURE	-40°C	-45°C
MAXIMUM FUEL TEMPERATURE	+57°C	**
MAXIMUM ALTITUDE	45,000 FEET	**
MAXIMUM ASYMMETRIC FUEL DIFFERENTIAL FOR NORMAL OPS.	400 POUNDS	400 POUNDS
EMERGENCY ASYMMETRIC FUEL DIIFFERENTIAL *	800 POUNDS	800 POUNDS

* Maximum lateral fuel imbalance is 400 lbs. A lateral fuel imbalance of 800 lbs has been demonstrated for emergency return.

** Refer to Figure 2-5A, Section II of the *AFM*

QUESTIONS

1. In the event of a loss of main DC power while operating in crossfeed:
   A. The crossfeed valve will fail closed.
   B. Crossfeed will continue.
   C. The LO FUEL PRESS L or R annunciator will illuminate.
   D. The motive flow valve for the receiving side will fail "open" and X-feed will terminate.

2. During initial engine starting, the primary source of fuel pressure to the engine-driven pump is:
   A. Motive flow fuel pressure.
   B. Primary ejector pump pressure.
   C. Respective side electric boost pump pressure.
   D. Suction pressure from the engine driven pump.

3. The primary ejector fuel pump:
   A. Provides motive flow fuel pressure.
   B. Provides head pressure to the engine-driven fuel pump.
   C. Provides high pressure, low volume fuel to the engine-driven fuel pump.
   D. Is located in the surge tank.

4. During initial engine start, the electric boost pump is activated when the:
   A. Start button is depressed.
   B. Throttle is advanced from cutoff to idle.
   C. Placing the boost pump switch to ON.
   D. Fuel low pressure switch.

5. During over-the-wing fueling:
   A. Fill the wing tanks until fuel fills the standpipe.
   B. It is not necessary to ground the refueling apparatus.
   C. Fill the wing tanks until fuel reaches the bottom of the standpipe.
   D. None of the above.

6. Select the correct choice regarding Single Point Pressure Refueling:
   A. Immediately after fuel flow has stabilized, perform a precheck test.
   B. A fuel flow precheck test is not required if a partial load of fuel is desired.
   C. Extreme care must be observed when attaching the fueling nozzle in order not to spill fuel.
   D. The refueling/defueling compartment is located directly forward of the left wing.

7. Opening a defuel select lever:
   A. Allows defueling the corresponding wing tank.
   B. Prevents defueling the opposite wing tank.
   C. Prevents refueling the corresponding wing tank.
   D. Prevents defueling the corresponding wing tank.

CHAPTER 6
AUXILIARY POWER UNIT

CONTENTS

ILLUSTRATIONS

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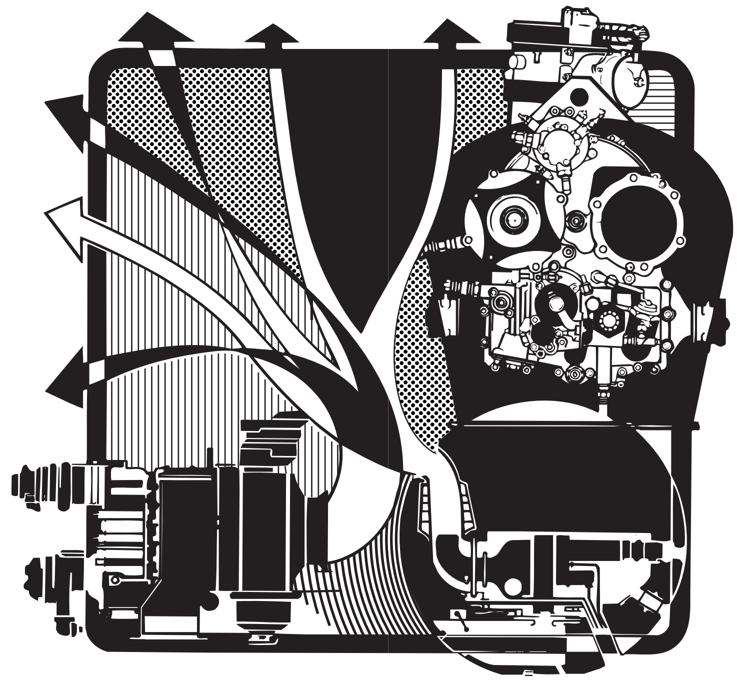
CHAPTER 6
AUXILIARY POWER UNIT



INTRODUCTION

The optional Auxiliary Power Unit (APU) installed on Citation Excel airplanes is a turbine-powered, Allied Signal Model RE100 (XL) unit. This chapter contains information useful to crew members about the APU system control and operation.

NOTE

If an optional APU is installed, the standard vapor cycle air conditioner will be removed.

APU installations will increase the basic empty weight of the airplane by approximately 105 lbs plus ballast, if required.

GENERAL

The APU provides supplemental electrical power and high-pressure bleed air for the airplane. The APU may be used on the ground or in flight as a source of bleed air for the airplane environmental and service air systems, and airplane electrical power requirements.

The APU starter-generator is rated at 28-volt, 300-ampere hours, but regulated not to exceed

230 amps. It will provide electrical power to all airplane electrical systems. If engine generators are online, the APU generator will parallel with the engine-driven generators to share the total electrical load. It is also utilized as a starter to crank the APU during start initiations.

NOTE

The APU generator **cannot** be used as a physical replacement for an engine-driven generator.

The APU is certified for ground and in-flight use. Maximum altitude for starting is 20,000 feet, and during operation, 30,000 feet.

The right wing fuel tank is the normal fuel supply for the APU. However, the left wing fuel tank can supply fuel during fuel crossfeed operation.

The APU is equipped with self-contained oil, ignition and fuel control systems. During starting and operation, the APU is monitored by an Electronic Control Unit (ECU).

NOTE

The APU is not certified for unattended use.

Separate fire protection, consisting of monitored fire detection and automatic fire-extinguishing system is provided for the APU.

The APU requires main DC power from the airplane's electrical system and sustained fuel pressure from a wing fuel tank electric boost pump (normally the right wing tank) for start and operation.

DESCRIPTION

The APU is mounted in a light-weight titanium steel fireproof enclosure in the upper aft right side of the tailcone compartment (Figure 6-1). The APU is a fully automatic, single-shaft, constant speed gas turbine engine (Figure 6-2). It can be accessed from the fuselage through an external panel above the right engine pylon (Figure 6-3). It utilizes a single-stage centrifugal impeller and a single stage radial turbine, mounted on a common shaft (power section). The gearbox driven by the power section, reduces the high speed power section rpm to lower speed required to drive the DC starter-generator, oil pump and the fuel control unit. The gearbox contains the integral oil reservoir.

The APU requires only electrical power, fuel, and control signals from the airplane for operation.

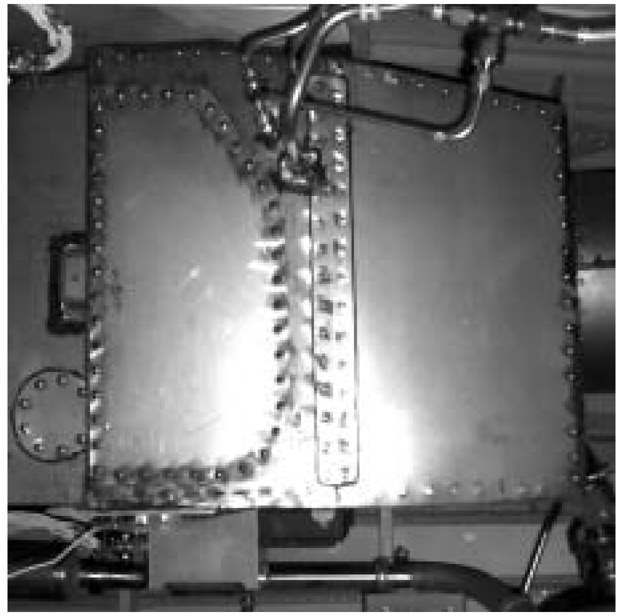


Figure 6-1 APU Enclosure, Tail Cone

Figure 6-2 APU Allied Signal Model RE100 (XL)

Figure 6-3 APU Access Panel

The Electronic Control Unit (ECU) automatically controls the APU to maintain required rpm and safe Exhaust Gas Temperature (EGT) throughout its operating range, including starts. The ECU monitors selected parameters of operation and if any are exceeded, the ECU will automatically shutdown the APU. Major APU parameters can be monitored in the cockpit from the control panel located on the subpanel directly in front of the copilot's circuit breaker panel (Figure 6-4).

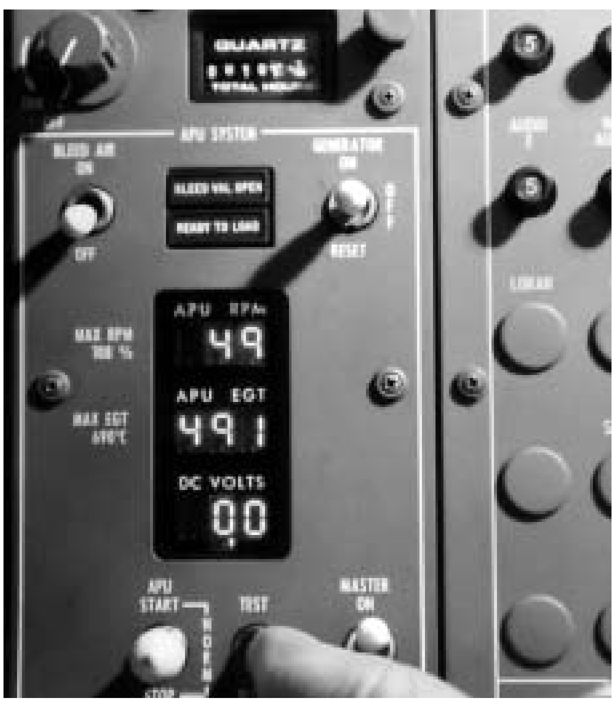


Figure 6-4 APU Control Panel — Test

CONTROLS AND INDICATORS

The APU control switches, RPM , EGT and DC VOLTAGE indicators, and two annunciator lights, are located on the control panel (Figure 6-4). The upper annunciator is labeled BLEED VAL OPEN, and the lower annunciator is labeled READY TO LOAD and both illuminate "white."

Additional monitoring of the APU is available from annunciators located on the copilot's instrument panel above the PFD controller (Figure 6-5). The three annunciators are: "red" APU FIRE switchlight, "amber" APU RELAY ENGAGED light, and an "amber" APU FAIL light.

An APU ammeter is located near the OXYGEN gage on the copilot's instrument panel to monitor APU generator loads (Figure 6-6).

An APU maintenance panel is mounted in the tailcone compartment directly above the light switch (Figure 6-7). The maintenance panel provides oil level checks during preflight inspections, and provides an emergency shutdown independent of the cockpit APU control panel.

Figure 6-5 APU Annunciators Copilot's Instrument Panel

Figure 6-6 APU Ammeter Copilot's Instrument Panel

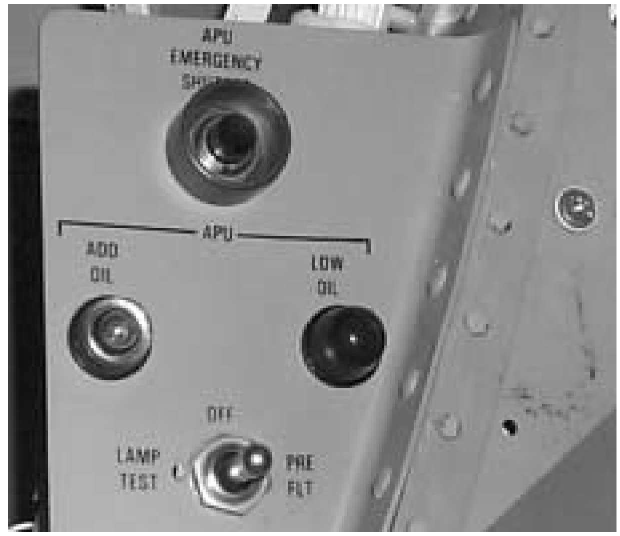


Figure 6-7 APU Maintenance Panel

The APU control panel, maintenance panel and associated indicators will be discussed in detail under OPERATION below.

SYSTEMS OVERVIEW

The APU is divided into three main sections:

- Compressor section
- Turbine section
- Gearbox

The compressor section consists of:

- Air inlet duct
- Inlet housing and screen
- Single-stage centrifugal compressor impeller
- Deswirl deflector

The turbine section consists of:

- Turbine housing (plenum)
- Annular reverse flow combustor
- Turbine nozzle
- Single-stage radial in-flow turbine motor

The gearbox section provides mounting for the:

- Starter/generator
- Oil pump
- Fuel Control Unit (FCU)

Power is developed through compression of ambient air by the centrifugal compressor. The compressed air is mixed with fuel, ignited and directed through the turbine nozzle assembly to drive the single-stage turbine. The turbine rotor, mounted on a common shaft, drives the compressor and the gear box. The gearbox reduces high-speed low-torque shaft power to low-speed high-torque power required to drive the APU accessories. It also houses the oil reservoir.

The gas turbine is started by the electrical starter/generator that rotates the turbine shaft through the accessory gear train. After the unit is started, the compressor draws ambient air through the APU air intake on the fuselage above the right engine pylon (Figure 6-8).

The compressor then delivers air under pressure to the combustor via the deswirl deflector. Fuel is mixed with high pressure air in the combustor and ignited. Air temperature and volume are greatly increased in the combustor by the

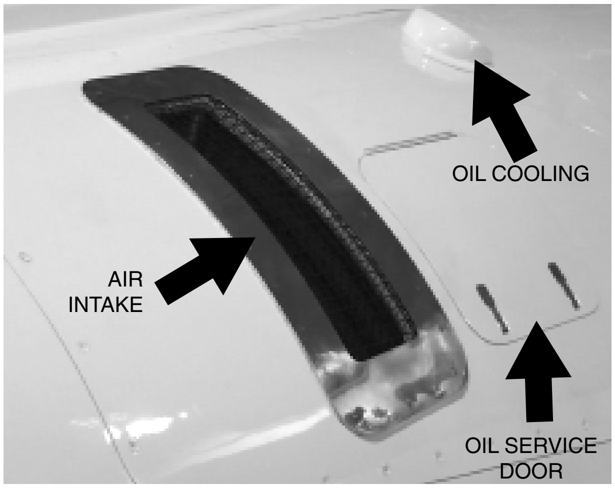


Figure 6-8 APU Compressor Air Intake Upper Right Fuselage

burning mixture. Hot gases increase in velocity through the exhaust nozzle and impinge on the turbine wheel. The turbine converts thermal energy into shaft power which in turn rotates the high-speed compressor (70,200 rpm at 100%) and the accessory gearbox. Turbine gases exhaust overboard through an opening in the fuselage above the right engine pylon aft of the APU air intake (Figure 6-9).

The accessory gear box drives the components necessary for APU operation and control.

LOAD REQUIREMENTS

The APU provides two types of power for the aircraft. Shaft horsepower via the gearbox mount pad to drive the auxiliary generator and pneumatic power (bleed air) to operate the aircraft environmental and service air systems.

If both types of power are demanded, shaft horsepower will have priority. All load requirements are controlled by selector switches on the APU control panel (Figure 6-4). When the APU is operating at 100% rpm with no load requirement (electrical or bleed air), the APU is at “idle” power.

When loads are demanded, the Electronic Control Unit (ECU) adjusts fuel flow

Figure 6-9 APU Turbine Exhaust Upper Right Fuselage

automatically to maintain constant 100% speed. When the APU generator is placed ON, a load is applied to the APU which initially causes the rpm to drop. The ECU responds by signaling the Fuel Control Unit (FCU) to increase fuel flow. The increased fuel flow will return the speed to 100%. Since the APU is controlled electronically, rpm drop is minimal and recovery is instantaneous and seldom seen on the APU rpm indicator (Figure 6-4). Shaft loads alone will not cause a very high EGT since all air being moved by the compressor is available to mix with and cool the combustion gases.

The APU is designed to provide bleed air into the aircraft air duct system by diverting some compressor discharge air from the combustor path. When APU bleed air is turned ON, air is diverted into the aircraft duct system and is no longer available to help drive or cool the turbine wheel. This will result in a much higher EGT. The two factors that influence EGT under bleed load conditions are aircraft demand for bleed air and ambient air temperature and pressure. The APU overall design dictates that shaft loads shall have priority. If the EGT operating temperature limit is reached under dual load condition, the volume of bleed air extraction will be reduced automatically by action of the ECU and the Bleed Air Valve (BAV) to guarantee required shaft horsepower output.

ELECTRONIC CONTROL UNIT (ECU)

ECU Functions:

- Prestart Built In Test Equipment (BITE)
- Automatic Start Control
- Speed Control
- Protective Shutdown Capability
- Start Inhibit Capability
- Fault Storage
- Fault Reporting to the Field Service Monitor (FSM)

The Electronic Control Unit (ECU) is located on the right hand side of the APU containment box (Figure 6-10). The ECU front panel connecters allow the ECU to electronically interface with the APU and the aircraft. A Field Service Monitor (FSM) software program can also be connected to the ECU for maintenance troubleshooting and downloading maintenance history data.

The ECU controls APU operation by receiving sensor inputs and commands, and sending control outputs to the aircraft systems and APU subsystems.

A prestart test is initiated, after the MASTER switch is placed ON, by depressing the TEST button on the APU control panel. If a failure is detected that would critically affect APU operation, the ECU inhibits starting and the failure is stored in the ECU and the APU FAIL annunciator will illuminate (Figure 6-5).

Figure 6-10 ECU and GCU

After a valid test is completed, the APU is started by placing the APU start switch to APU START. The ECU is alerted and controls the APU start automatically. The ECU controls ignition and fuel automatically as required for ambient conditions. Speed signals provide switch points for automatic fuel, ignition, starter relay dropout and load circuit arming.

Exhaust Gas Temperature (EGT) and engine speed are monitored continuously by the ECU and speed is controlled at full-rated speed, 70,200 rpm (100%).

The ECU regulates APU bleed-air output by sensing EGT and comparing it to a predetermined schedule within the ECU. If an overtemperature is likely to occur, the ECU will reduce bleed-air extraction.

The ECU will automatically shutdown the APU if speed exceeds 108% or EGT exceeds scheduled limits as determined by ambient conditions. The ECU monitors specified parameters and has the authority to shut down the APU if continued operation might cause damage.

FUEL SYSTEM

APU fuel is normally extracted from the right engine fuel manifold supplied from the right wing fuel tank (APU fuel line is connected prior to the right engine fuel shutoff valve). Anytime the the APU is started or operating, the right fuel boost pump will automatically activate provided the R–FUEL BOOST switch is in NORM. The LO FUEL PRESS–R annunciator will be extinguished during APU start and operation. The APU requires positive fuel boost pump pressure for starting and operation. The right boost pump will terminate when the APU is shutdown.

NOTE

When the right boost pump is commanded ON, only for APU operation, the FUEL BOOST–R annunciator will not illuminate. However, if the APU is running and the pump is commanded ON for any other reason (crossfeed, low fuel pressure, main engine start, etc.) the FUEL BOOST–R light will illuminate.

Crossfeed

If fuel crossfeed is desired from the left wing tank while the APU is operating, the right fuel boost pump will automatically shut off to allow fuel transfer. Selecting fuel crossfeed OFF, the right fuel boost pump will automatically reactivate after the system cycles OFF.

Selecting crossfeed from the right wing tank while the APU is operating, the right boost pump will continue to operate and the FUEL BOOST–R annunciator will illuminate. Terminating crossfeed, the FUEL BOOST–R light will extinguish (boost pump continues to operate).

NOTE

Fuel boost pump operation is not available for the APU if the right boost pump switch is OFF.

If prolonged use of the APU results in a fuel imbalance between the left and right wing tanks, selecting crossfeed from L TANK to R ENG should correct the imbalance after a period of time.

Fuel Shutoff

An airframe-mounted solenoid-operated fuel shutoff valve is installed in the fuel supply line to the APU (outside of the APU enclosure). It is energized open automatically when the APU is started and remains open while the APU is operating. The valve deenergizes closed when the APU is shut down normally, or during emergency shutdowns, and when the APU FIRE switchlight is illuminated.

NOTE

After the APU is operating "loaded," on the ground or in flight, approximately 110 pph fuel flow should be reflected in the fuel section of the FMS.

Fuel Flow

The APU fuel supply is pressurized by an aircraft electric fuel boost pump. Fuel is directed to the integral FCU/high-pressure fuel pump, where it is filtered, and then flows through a metering valve that is commanded by the ECU based on engine power requirements. Fuel then passes through an "internal" APU fuel shutoff solenoid (energized open at 5% rpm), allowing unrestricted fuel flow to the primary manifold during the initial start phase (Figure 6-11). The fuel solenoid valve will remain open until the ECU removes electrical power due to a normal or protective shutdown. As fuel pressure increases with rpm acceleration, the flow divider will open and also allow fuel to the secondary manifold (both manifolds charged during normal operations).

The primary fuel manifold provides high-pressure fuel to two primary nozzles and the secondary manifold supplies high-pressure fuel to four secondary nozzles. The nozzles are distributed evenly around the turbine plenum and protrude into the combustion chamber. The nozzles provide proper fuel atomization for initial combustion and maximum energy extraction by the turbine.

A drain line extends through the fuselage on the bottom right side of the tailcone to drain any fluids that may accumulate including moisture that may collect from the openings above the right engine pylon during inclement weather conditions (Figure 6-12).

Fuel flow is controlled by the ECU as determined by acceleration schedules during starting, EGT monitoring, ambient air conditions (temperature and pressure), shaft loads and bleed-air demands by the aircraft.

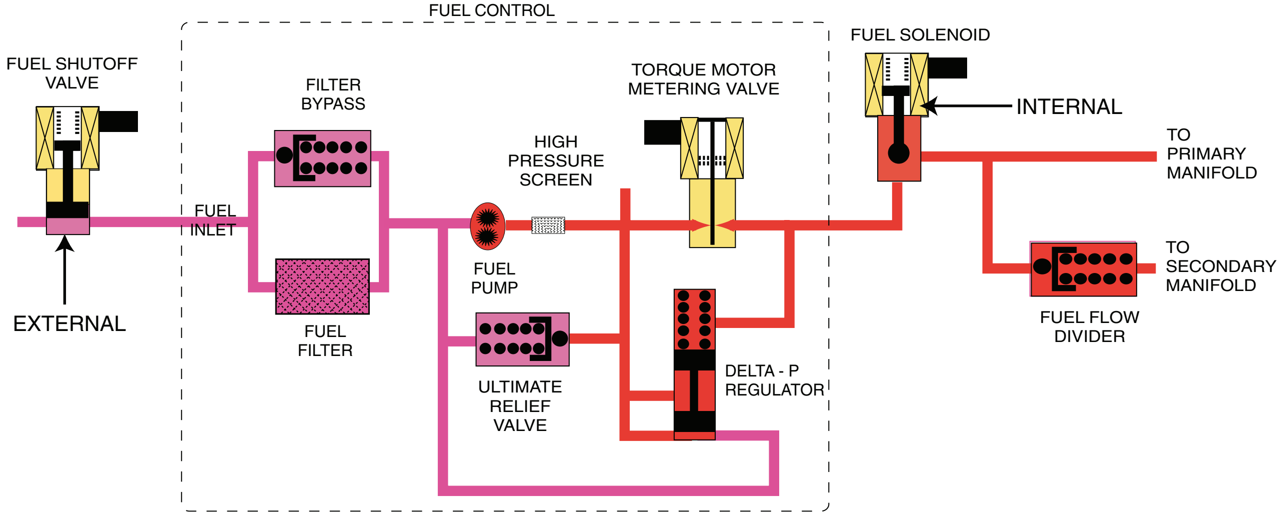


Figure 6-11 APU Fuel System Schematic

Figure 6-12 APU Drain Line

LUBRICATION SYSTEM

The lower portion of the gearbox is cast to form an oil reservoir. An oil temperature sensor is installed in the reservoir and signals the ECU for over-temperature conditions. A Low Oil Pressure (LOP) switch in installed in the oil pressure line (past the filter) to signal the ECU of low oil pressure indications (Figure 6-13).

The critical points requiring lubrication are supplied by a combination of drilled passages and a transfer manifold. Remaining areas that require lubrication are served by splash oil.

The rotating shaft is supported by two bearings, a forward ball bearing and an aft roller bearing. A carbon seal is installed behind the aft bearing to seal the oil-wetted area.

The oil system is cooled by two methods:

- The surface of the gearbox is cooled as air is pulled through the APU compartment from the ram air inlet on the access panel (Figure 6-8) and exhausted through the APU exhaust.
- Air flow through the inlet extracts heat from the oil by cooling the rear side of the gearbox.

The oil system components include:

- Oil pump
- Low Oil Pressure (LOP) switch
- Oil temperature sensor
- Gearbox vent
- Magnetic chip collector
- Oil level switch

Oil System Operation

The oil pump draws oil directly from the reservoir and forces oil under pressure through jets and passageways. Oil pressure will normally vary between 60-80 psig depending on oil temperature conditions. To prevent excessive system pressure, an ultimate pressure relief valve is set at 200 psig.

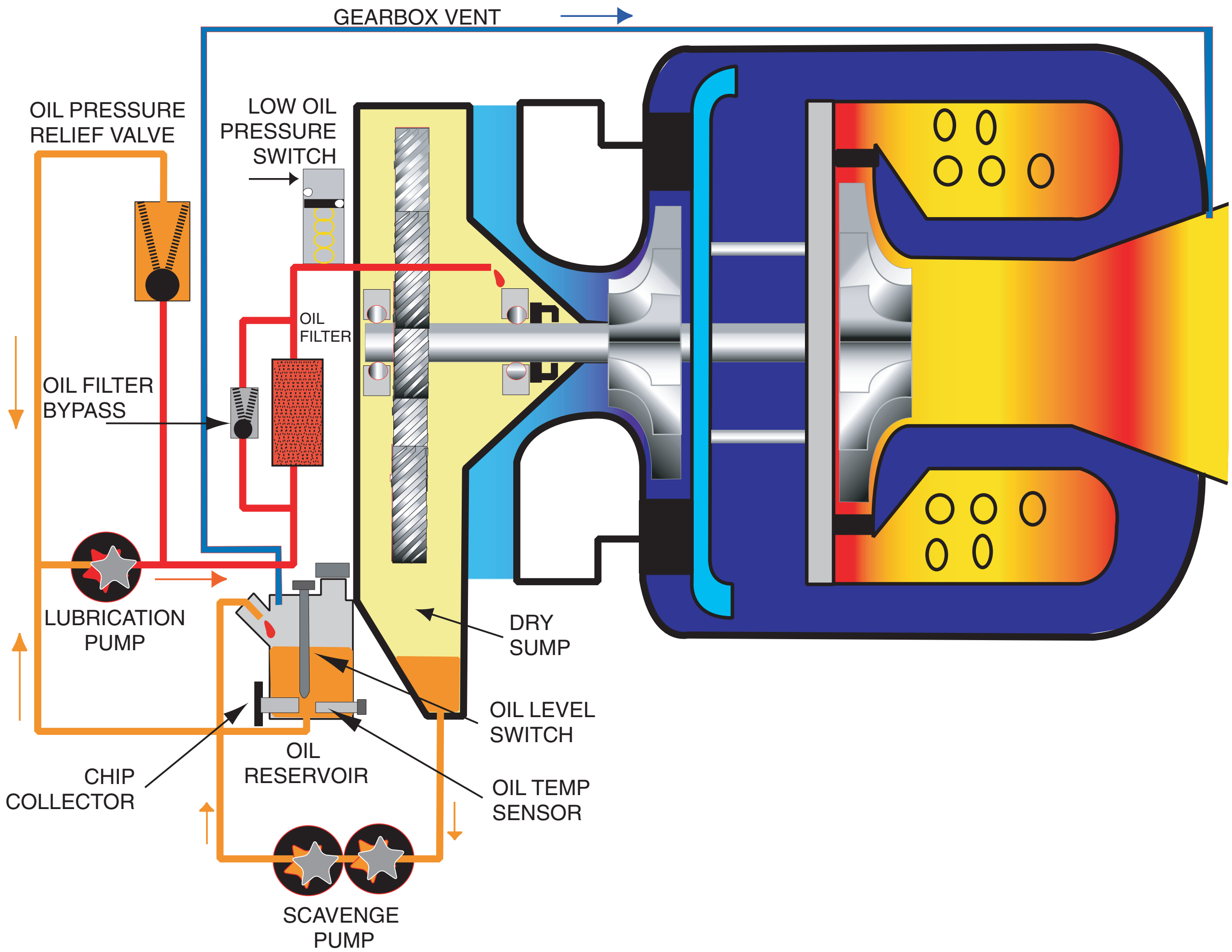


Figure 6-13 APU Oil System Schematic

Oil passes through a 10-micron filter which has a bypass provision. From the filter, oil passes through internal passageways and manifolds to the main shaft bearings, planetary gear system bearings, and the Fuel Control Unit (FCU) drive shaft. Oil is also directed to the low oil pressure switch (Figure 6-13).

Oil from various lubrication points is gravity drained to the sump area. As oil flows to the sump along the inner wall of the gear case housing, it is cooled by ram air and compressor inlet air acting on the outside walls of the gearbox. Oil is then pumped back to the reservoir by the scavenge pump.

Oil temperature in the reservoir and oil pressure in the pressure line is continually monitored by the ECU.

The oil reservoir is serviced through a screened filler cap located on the right side of the APU gearbox and is accessed through a panel on the fuselage above the right engine pylon (Figure 6-8).

Oil service capacity is approximately 1.5 US quarts and an oil level switch indicates low oil level (refer to APU MAINTENANCE PANEL later in this chapter).

Gearbox Vent

Continuous gearbox pressure build-up must be released to prevent internal damage. This is accomplished by a vent located on top of the gearbox and extends aft to the tail pipe. The scavenge pump draws air from the sump. As it pumps oil into the reservoir, the mixture of air and oil is drawn through an air/oil separator that extracts air from the oil. Once separation is completed, air is vented to the tailpipe and exhausted overboard.

Low Oil Pressure Switch (LOP)

The LOP switch in the pressure line provides protection against low oil pressure conditions. During prestart, the LOP switch is checked "closed" (low pressure) by the ECU. If the switch is "open" the ECU will inhibit the start and record the fault in memory. During APU starting, as oil pressure increases the LOP switch "opens" at 40 psig (normal pressure) and normal start acceleration should occur. If the APU is operating above 95% rpm and the LOP switch closes for 10 seconds (less than normal pressure), the ECU will initiate a protective shutdown.

Oil Temperature Sensor

The oil temperature sensor in the reservoir provides protection against hot oil conditions during APU operation. During prestart, the sensor is checked by the ECU for a temperature range of -73°C (-100°F) to 260°C (500°F). If the temperature is out of range, the ECU will inhibit a start. If the APU is operating above 95% rpm and oil temperature exceeds 149°C (300°F) for 10 seconds, the ECU will automatically initiate a protective shutdown.

Oil Level Switch

The oil level sensor provides remote indication of oil level. The oil level switch will provide a signal to the tailcone maintenance panel when the oil level requires servicing or is too low to operate the APU.

APU Maintenance Panel

The APU maintenance panel is located in the tailcone. It provides oil level checks and contains an EMERGENCY SHUTOFF button independent of the APU control panel in the cockpit. A LAMP TEST–OFF–PREFLT switch located below the oil lights, allow the crew to perform a preflight test of oil quantity (Figure 6-7).

Magnetic Chip Collector

The chip collector is located at the lowest point in the reservoir. The magnetic element is inspected periodically by maintenance personnel. If metal particles are detected upon inspection, the APU maintenance manual must be consulted to determine whether APU operation can be continued.

NOTE

There is no visual annunciation of metal detection in the cockpit.

Pneumatic System

The APU provides compressed "bleed air" from the compressor section to the airplane environmental and service air systems (Figure 6-14). Bleed air flows from the APU to the aircraft pneumatic ducts through the APU Bleed Air Valve (BAV). The BAV is controlled "open" and "closed" by the BLEED AIR switch on the APU control panel (ON–open, OFF–closed). Bleed air extraction is available at 95% rpm plus four seconds (READY TO LOAD light illuminates) (Figure 6-14).

Bleed Air Valve (BAV)

The BAV is mounted outside the APU enclosure forward of the exhaust duct in the tailcone (Figure 6-15). The BAV is an aircraft supplied component. The BAV controls bleed-air flow to the airplane environmental system and protects the APU from overtemperature conditions. The BAV is opened by placing the BLEED AIR switch ON (BLEED VAL OPEN light illuminates).

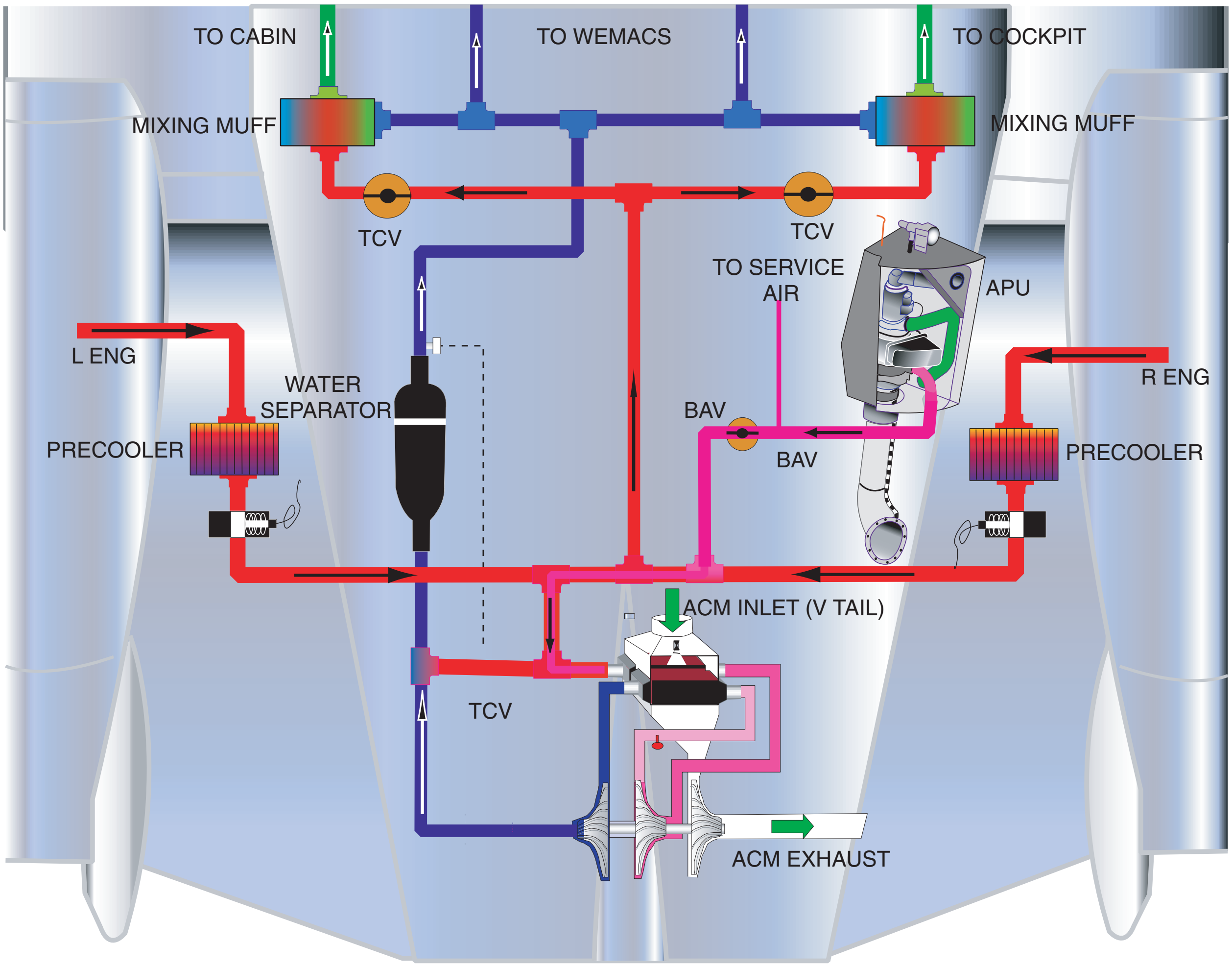


Figure 6-14 APU Bleed-Air Schematic

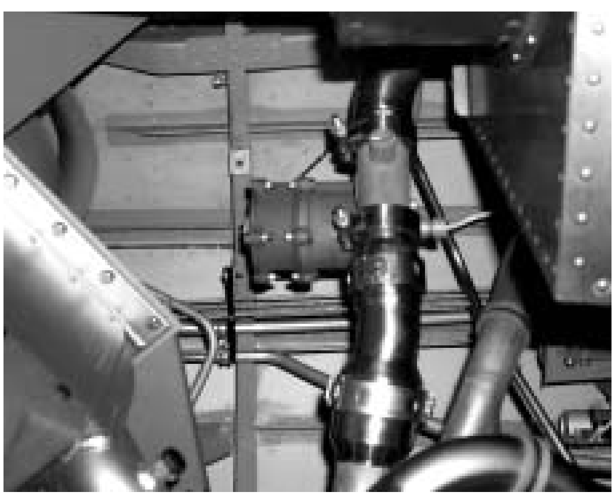


Figure 6-15 APU Bleed-Air Valve

Positioning the BLEED AIR switch ON, signals the ECU to send discrete signals to energize the BAV solenoid and allow a portion of APU compressor discharge air to push the BAV open. The BAV will initially open to a half-open position for 20 seconds before it moves to full open.

The ECU regulates the APU pneumatic output power by sensing Exhaust Gas Temperature (EGT) and inlet ambient air temperature. If EGT exceeds a preset temperature limit, the ECU signals the BAV to close to a half-open position. This action prevents the APU from reaching an overtemperature condition. The ECU continues to monitor EGT and will automatically signal the BAV to full open when the EGT falls below a preset limit.

NOTE

If APU bleed air is selected and EGT exceeds 649°C (1200°F) for four seconds, the ECU automatically signals the BAV to assume a one-half open position reducing the pneumatic load on the APU and allowing the EGT to lower. Once the EGT drops below 482°C (900°F) for 20 seconds, the ECU signals the BAV to a full open position.

NOTE

If EGT reaches 690°C (1275°F), the ECU will automatically shut down the APU.

If APU bleed air is no longer required, or prior to a normal APU shutdown, the BLEED AIR switch is placed OFF. The ECU removes electrical power from the BAV solenoid and the BAV is programmed to travel fully closed in one to three seconds to prevent abrupt APU load transitions and airplane environmental "bumps."

ACM Overheat

If the airplane Air Cycle Machine (ACM) experiences an overheat condition (ACM O'HEAT annunciator illuminated), the ECU will signal the BAV to close until the overheat condition is corrected.

Electrical System

Electrical accessories/components are used in conjunction with the Electronic Control Unit (ECU) to assist in performing sensing and control functions to start and monitor the APU.

Starter/Generator

The starter/generator is mounted on the front of the gearbox. It provides starting power to rotate the APU and supplies supplemental DC power for the aircraft electrical system.

The APU generator switch on the control panel is used to place the generator online and provide reset capability.

Placing the generator switch ON, allows the APU generator relay to close connecting the APU generator to the airplane crossfeed bus (Figure 6-16). The APU ammeter will then indicate a load.

Ignition System

The ignition system consists of an ignition unit, ignition lead and an ignition plug that extends into the combustor.

The ignition system is fully automatic and controlled by the ECU. During start, at 5% rpm, the ignition unit is energized. At 99% rpm ignition is deenergized. Should a flameout occur during operation, an "Auto Relight" function will occur at 94% rpm.

Speed Sensor

The speed sensor, located on top of the gearbox, provides speed signals to the ECU to control APU operation. The sensor provides speed signals to the ECU to sequence the following events:

- Initiation of fuel flow (5% rpm)
- Ignition on (5% rpm–ON; 94% rpm–Auto Relight)
- Starter cutout (50% rpm; 60% rpm–Backup)
- Acceleration fuel schedule
- Ignition termination (99% rpm)
- Generator loading (95% rpm + 4 sec)
- Ready-to-load operation (95% rpm + 4 sec)
- Overspeed protection (108% rpm)

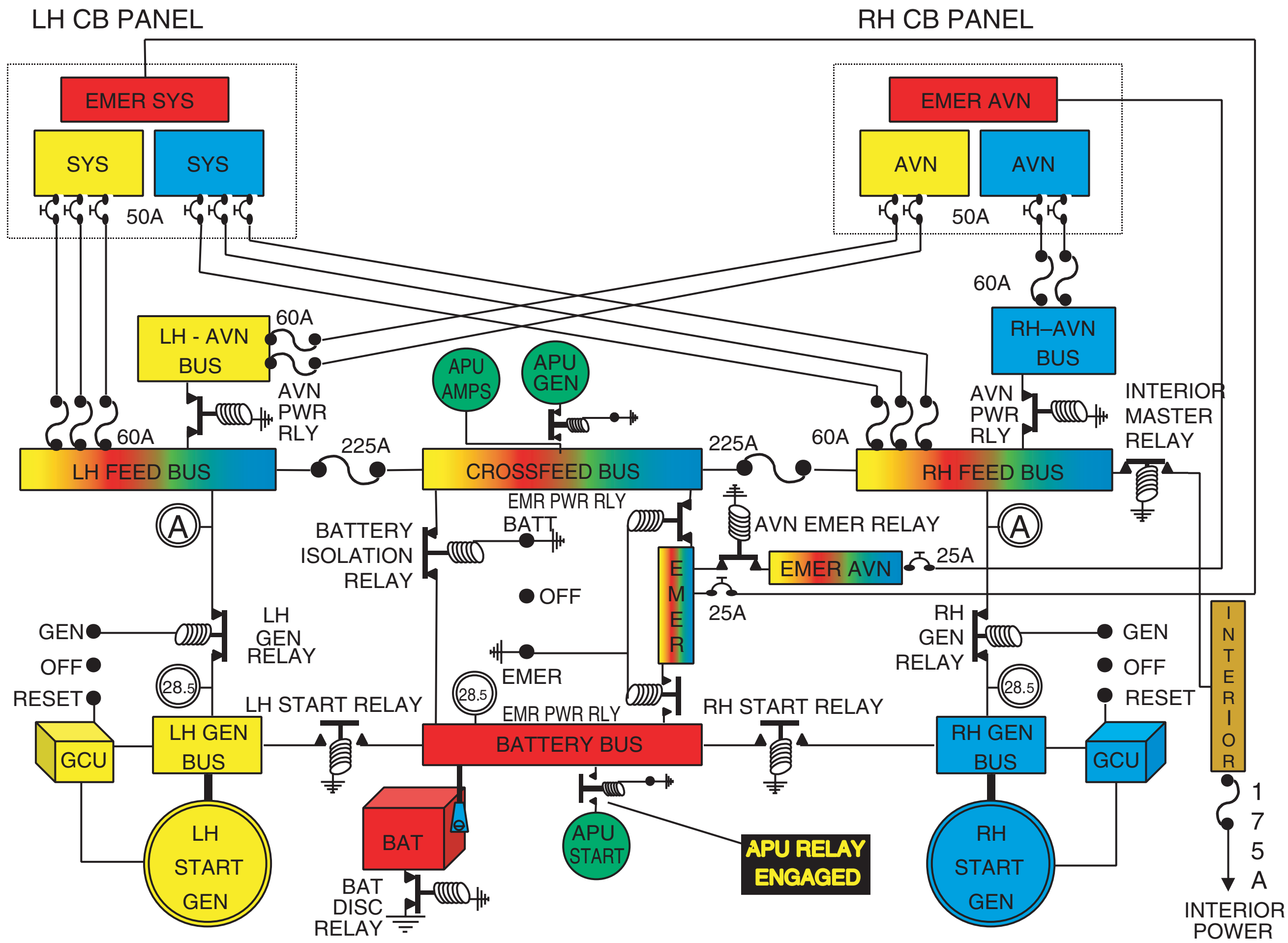


Figure 6-16 Aircraft and APU Electrical Schematic

Exhaust Gas Thermocouple (EGT)

The EGT thermocouple is located at the 9:00 o'clock position on the aft section of the exhaust pipe. The thermocouple provides EGT temperature input to the ECU in order to control fuel output based on acceleration and prevent over-temperature conditions. The thermocouple also provides input to the EGT gauge on the control panel.

Continuous EGT input to the ECU is required for all phases of APU operation. During pre-start, the ECU checks the thermocouple for an input range of-73°C (-100°F) to 1093°C (2000°F). If the temperature is out of this range, the ECU will inhibit the start and log the fault in memory.

The ECU receives EGT signals to:

- Indicate EGT
- Maintain fuel trim schedules
- Control the Bleed Air Valve (BAV)
- Trigger APU shutdowns if EGT limits are exceeded

Inlet Temperature Sensor (T_2)

The inlet temperature (T_2) sensor monitors APU inlet temperature for fuel scheduling limits, reverse flow protection, and no flame detection limits. The ECU monitors T_2 temperature continuously to update schedule limitations.

When the APU is operating with the BAV open, the ECU checks T_2 temperature for a range of -73°C (-100°F) to 93°C (200°F). If temperature is out of range, the ECU executes a protective shutdown of the APU.

Start Counter

The APU start counter is mounted on the left side of the APU between the ignition unit and the fuel solenoid valve, on the forward side of the plenum. The start counter retains total APU start cycles.

Fire Protection

Fire protection is provided by a fire detector, a halon-charged fire-extinguisher bottle, and an associated warning switchlight. A continuous fire-detection loop inside the APU enclosure contains a fixed charge of inert gas. Fire detection is activated by a pressure sensor that activates as inert gas pressure in the fire-detection loop is increased by heat. The sensor activates the "red" APU FIRE light "steady" on the copilot's instrument panel.

NOTE

The airplane MASTER WARNING annunciators will not illuminate if the APU FIRE light illuminates.

The TEST button on the APU control panel checks integrity of the entire fire-detection loop, including adequate gas pressure and condition of the sensor. Depressing the TEST button causes the APU FIRE light to illuminate if the detection system is functional.

NOTE

The APU fire detection and extinguishing system, including the TEST button, requires main DC electrical power.

A fire-extinguisher bottle, separate from the main engine fire bottles, is provided for APU fire suppression (Figure 6-17). It is configured with a temperature and pressure-compensated switch that will illuminate the APU FAIL light on the copilot's instrument panel if bottle pressure is low or the bottle is empty. An APU FAIL light illuminated prior to start (MASTER switch ON) will inhibit the start.

Figure 6-17 APU Fire-Extinguisher Bottle

The fire bottle is plumbed into the APU steel enclosure. In the event of fire (APU FIRE light illuminates) the APU will immediately shut down. If the APU FIRE switchlight is not depressed within eight seconds of continuous illumination, the fire bottle will discharge automatically.

The crew may activate the fire bottle manually by lifting the guard and depressing the APU FIRE switchlight within eight seconds of continuous illumination.

NOTE

The APU FAIL light will illuminate and remain illuminated after the fire extinguisher is discharged to indicate low bottle pressure.

When the APU FIRE light illuminates, the APU fire system automatically initiates the following:

- Signals the ECU to initiate an immediate shutdown.
- Trips the starter/generator field to prevent reignition.
- Deenergizes the APU aircraft-mounted fuel shut-off valve.
- Deenergizes the right electric fuel boost pump.
- Logs APU fire protective shut down into ECU memory.
- Eight second time delay before the APU fire extinguisher bottle agent is deployed.

The APU fire system resets when a fire is no longer present.

OPERATION

The APU is interfaced from the cockpit through the APU control panel via the Electronic Control Unit (ECU). The control panel provides the following information:

- Master power
- Normal start and stop sequencing
- Lamp test
- Generator power
- Generator voltage indication
- Bleed air valve power
- Ready to load indication
- RPM indication (%)
- EGT indication (°C)

APU CONTROL PANEL FUNCTIONS

MASTER Switch

Placing the MASTER switch ON, powers up the ECU and provides power for the control panel gages, switches and annunciators.

START–STOP Switch

The START–NORM–STOP switch alerts the ECU to initiate the start sequence and enables on-speed operation. STOP position initiates a simulated overspeed signal to cause a normal shutdown.

TEST Button

Depressing the TEST button performs a lamp test of all the annunciator lamps on the APU panel and the copilot's instrument panel. The test function also tests the digital indicators and the fire-detection system.

GENERATOR Switch

The GENERATOR ON–OFF–RESET switch allows the APU generator to be connected to the crossfeed bus after the APU is running. If the APU generator trips off line, the RESET position provides the capability to reset the generator if the problem that caused the trip is no longer a factor (resets the generator field relay).

BLEED AIR Switch

The BLEED AIR ON–OFF switch is used to open and close the Bleed Air Valve (BAV) once the APU is running. An open BAV allows APU bleed air into the aircraft environmental duct work and the service-air system.

BLEED VAL OPEN Lamp

The BLEED VAL OPEN light illuminates any time the Bleed Air Valve (BAV) is energized open.

READY TO LOAD Lamp

The READY TO LOAD light illuminates when the APU rpm is at or above 95% rpm for at least four seconds. It alerts the crew that the APU may be electrically-and/or pneumatically-loaded (generator on line and/or BAV open).

APU RPM Indicator

The digital rpm indicator indicates APU speed in percent (%).

APU EGT Indicator

The digital EGT indicator indicates APU Exhaust Gas Temperature (EGT) in °C.

DC Voltage Indicator

The digital voltage indicator, indicates APU generator voltage.

APU INDICATORS, COPILOT'S INSTRUMENT PANEL

APU Fire Switchlight

If an APU fire is detected in the APU enclosure, the "red" APU FIRE switchlight illuminates "steady" and the APU will shutdown immediately. The APU FIRE switchlight may be manually depressed to activate the APU fire extinguisher. If the fire switch is not manually activated, after an eight second delay the ECU will automatically fire the extinguisher bottle and deploy fire-extinguishing agent into the APU enclosure.

NOTE

The fire extinguishing agent is not harmful to the APU (same agent as the engine fire system).

APU RELAY ENGAGED Annunciator

The "amber" APU RELAY ENGAGED annunciator illuminates to indicate the APU start relay is energized closed during APU starts, or engine starts using the APU generator, allowing electrical power to or from the aircraft battery bus (Figure 6-5). The light should extinguish at 50% rpm during APU start acceleration.

APU FAIL Indicator

The "amber" APU FAIL annunciator illuminates when the ECU initiates an APU protective shutdown or detects a fault prior to start (inhibits start function until the fault is cleared) (Figure 6-5).

NOTE

The APU FAIL light will illuminate if the APU fire-extinguisher bottle pressure is low and inhibits the start.

APU MAINTENANCE PANEL

The APU maintenance panel located in the tailcone provides APU oil level checks during preflight exterior inspections and provides emergency shutdown capability independent of the APU control panel in the cockpit.

Depressing the APU EMERGENCY SHUTOFF button will remove electrical power from the ECU and cause the APU to shutdown.

The APU oil level is checked as follows:

- Lamp test is performed by placing the LAMP TEST–OFF–PRE FLT switch to LAMP TEST and observe that both the "amber" ADD OIL and "red" LOW OIL lights illuminate (bulbs are OK).
- If neither light illuminates during the PRE FLT check, the oil level is within normal operating range.
- If placing the switch to PRE FLT illuminates the "amber" ADD OIL light, the oil reservoir requires servicing. The APU may be operated and servicing may be delayed until the next available opportunity.
- If the "red" LOW OIL light illuminates, the APU may not be operated until the reservoir is serviced.

NOTE

The APU maintenance panel is powered from the airplane battery bus (hot).

APU START SEQUENCE

Prior to APU starting, the aircraft battery switch must be ON.

NOTE

APU starts on the ground may be aircraft battery starts only (Figure 6-18), GPU starts only (battery disconnect relay opens during start), or aircraft generator(s) assisted battery starts (Figure 6-19).

NOTE

In flight APU starts are battery only starts (squat switch logic prevents generator assisted APU starts). In-flight starts are prohibited above 20,000 feet (Figure 6-20).

MASTER Switch — ON

The MASTER switch is placed ON to provide electrical power to the ECU. The ECU performs APU power-up tests. After the power-up tests are completed, the ECU accomplishes the prestart Built-In-Test Equipment (BITE) test to ensure that no faults exist that would inhibit a start. If a fault is detected, the APU FAIL light illuminates. Placing the MASTER switch ON,

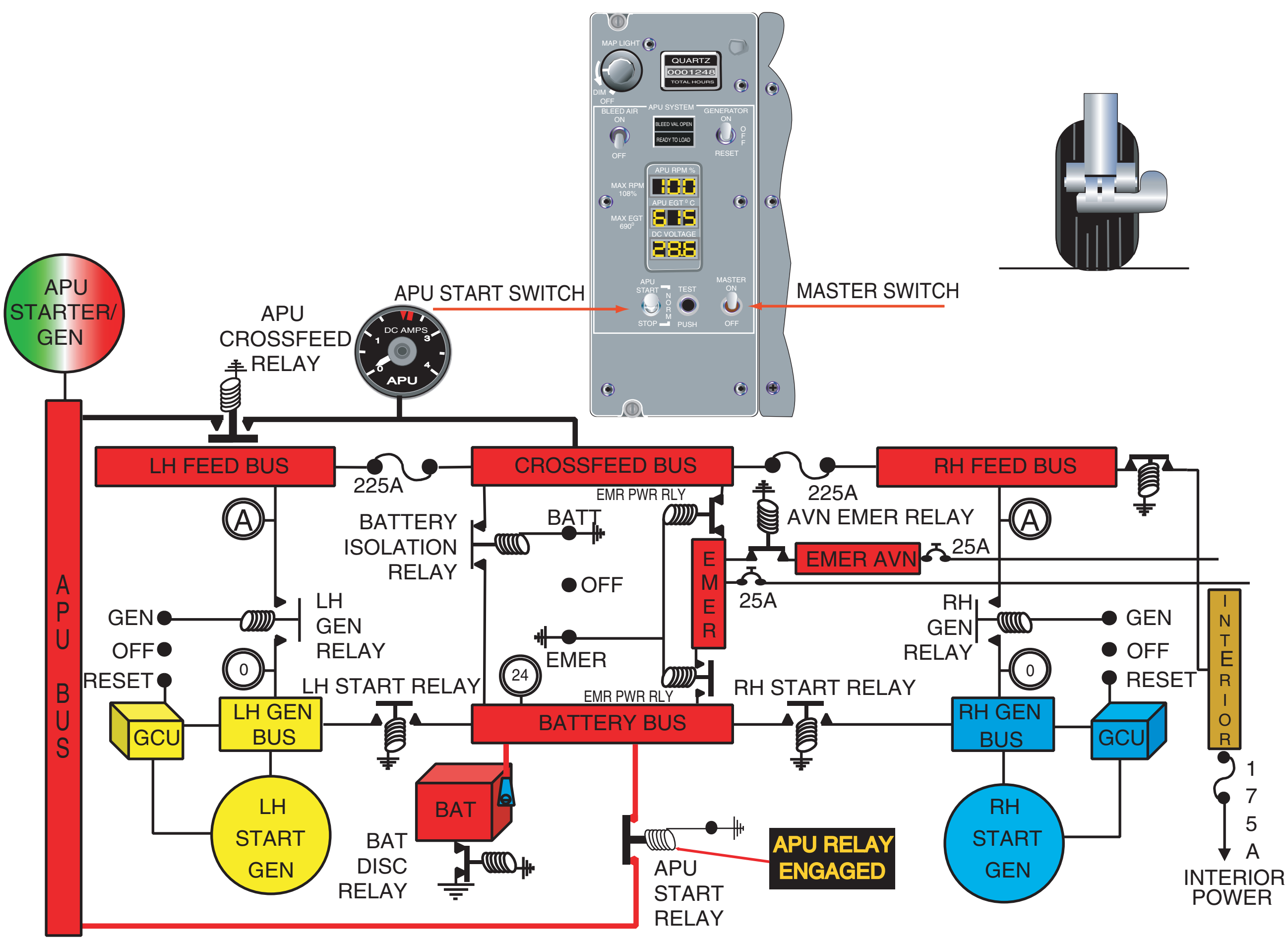


Figure 6-18 APU Start — On Ground (Engine Generators Off Line)

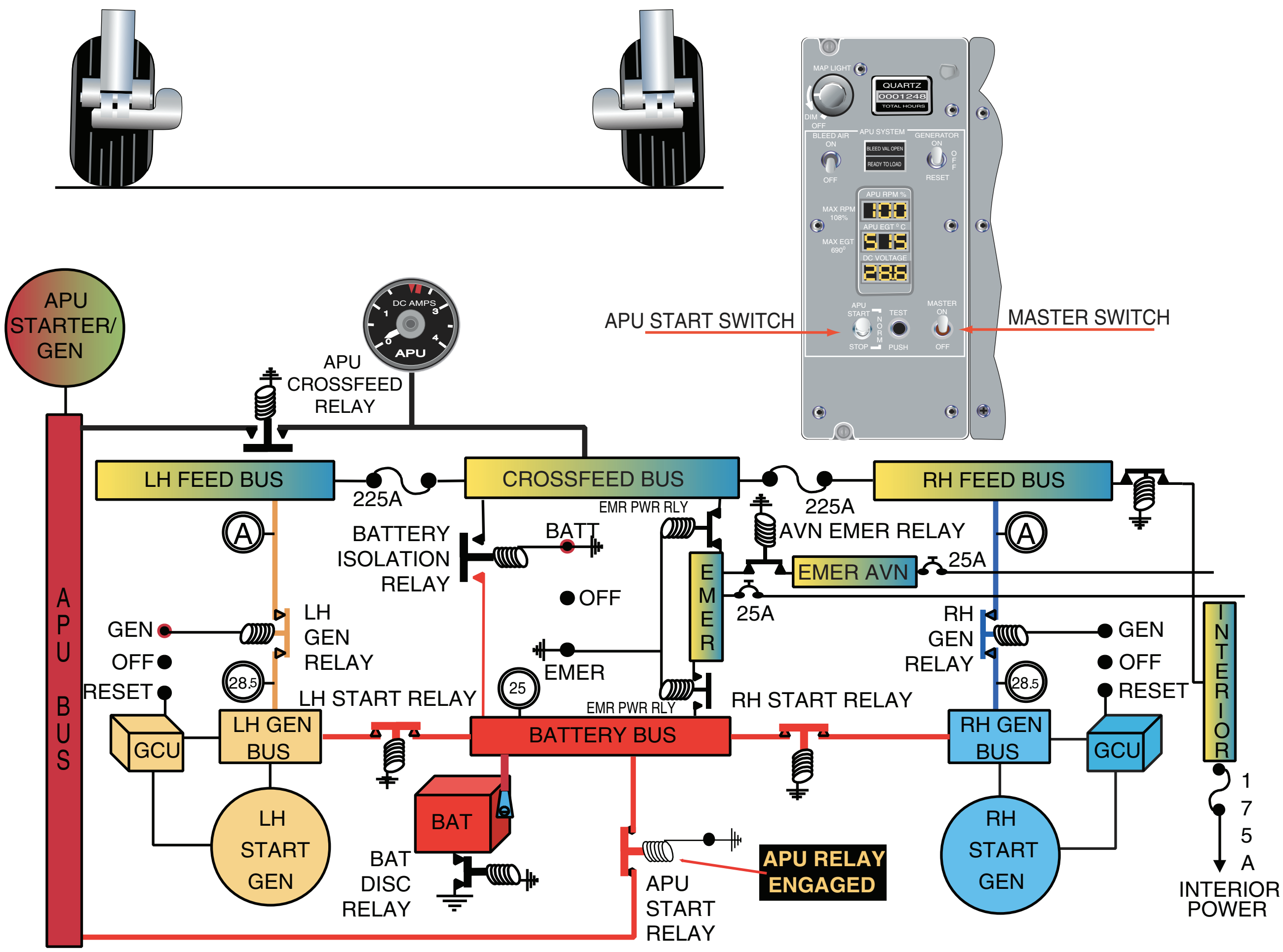


Figure 6-19 APU Start On Ground (Engine Gen(s) On Line and Battery Assist)

the APU digital indicators should stabilize and indicate:

- APU RPM % — “0”
- APU EGT °C — “0.00”
- DC VOLTAGE — “0”

APU TEST Button

A lamp test and circuit integrity is conducted prior to start by depressing the TEST button. The following lamps must illuminate during the test:

- APU FIRE (red), if fire detection circuit is valid.
- APU RELAY ENGAGED (amber)
- APU FAIL (amber)
- BLEED VAL OPEN (white)
- READY TO LOAD (white)

If a lamp or lamps do not illuminate during the test, an APU start must not be attempted until the condition is cleared.

NOTE

APU start attempt is prohibited when the APU FAIL light is illuminated (low fire bottle pressure or APU fault).

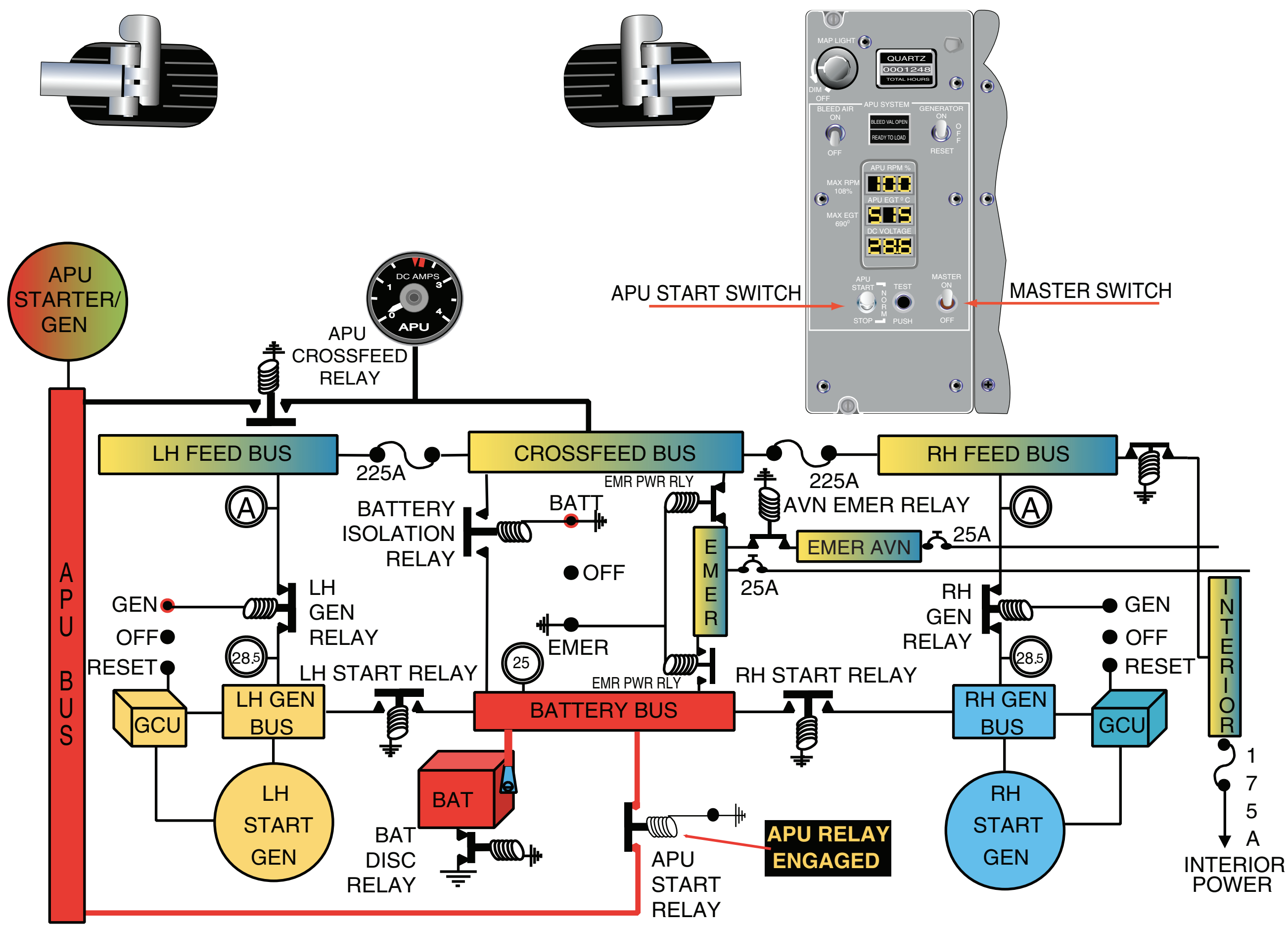


Figure 6-20 APU Start — In Flight

Digital indicators should display the following during test:

- RPM 49
- EGT .... 500 ±10
- DC VOLTAGE 00.0

Ensure BLEED AIR and GENERATOR switches are OFF.

APU START–NORM–STOP Switch–START

NOTE

Placing the APU START switch to START, alerts the ECU to provide automatic starting. The ECU controls ignition and fuel automatically during start as required for ambient conditions.

If no faults exist that would inhibit a start, the APU START–NORM–STOP switch is momentarily selected to START (spring-loaded to NORM). The aircraft right hand fuel boost pump activates (LO FUEL PRESS–R extinguishes if the RH engine is shutdown).

The APU Generator Control Unit (GCU) commands the start relay to energize closed, (annunciated by the APU RELAY ENGAGED light), and the starter begins motoring (rpm indicator digits increasing) (Figure 6-18).

NOTE

If the APU start is an engine generator(s) assisted start, the engine start relay(s) will close (engine start button light and/or lights will illuminate), and the APU start logic will command the battery isolation relay open and protect the 225-amp current limiters (Figure 6-19).

At 5% rpm, the ECU powers the ignition unit, fuel torque motor, and the APU fuel solenoid valve (open). During start, the ECU controls fuel scheduling through the primary and secondary fuel manifolds, and continually monitors engine speed and EGT limits as determined by ambient conditions (T_2).

If scheduled limits are exceeded, the ECU executes a precautionary shutdown (APU FAIL light illuminates). This is accomplished by removing power from the fuel solenoid, the ignition circuit and the GCU (opens the start relay). The fault code will be stored in memory for ease of maintenance during troubleshooting.

As the start sequence continues to accelerate, light-off will be indicated by a rise in EGT. The ECU monitors speed and EGT during acceleration to limit fuel output from the Fuel Control Unit (FCU).

The EGT shutdown limit during the initial start sequence to 50% speed, is 871°C (1600°F). As APU speed increases, the overtemperature limit is decreased proportionally as efficiency is increased. The EGT limit will vary from 871°C (1600°F) at 50% rpm to 690°C (1275°F) at 100% rpm.

At 50% speed, the speed sensor signals the GCU to deenergize the start relay and the APU RELAY ENGAGED light extinguishes.

NOTE

If the speed sensor fails and/or the GCU fails to open the start relay at 50%, the ECU backs up the GCU and opens the start relay at 60% rpm.

At 95% rpm the start counter records the start.

At 95% rpm plus four seconds, the ECU shifts to on-speed control. The READY TO LOAD light illuminates (start is complete). The APU may now be loaded electrically and pneumatically.

At 99% rpm, the ignition unit is deenergized.

At 100% rpm, the APU is considered on-speed. The ECU maintains constant rotor speed rpm at 100% ± 1.0% (70,200 rpm), and monitors EGT. The DC VOLTAGE indicator should display 28.5 VDC.

If APU speed drops below 94%, the ignition unit will automatically reenergize, unless the APU is in a protective or normal shutdown mode.

The programmed overspeed and EGT shutdown limits are established at 690°C (1275°F) and 108% respectively.

APU Loading

After the READY TO LOAD light illuminates, the APU generator may be placed on line. Placing the APU generator switch ON, energizes the APU generator power relay to connect APU generator output to the airplane crossfeed bus (Figure 6-16). The APU ammeter on the copilot's instrument panel should reflect an amperage load.

If desired, place the APU BLEED AIR switch ON to provide APU bleed air to the airplane environmental and service air systems. The ECU powers the Bleed Air Valve (BAV) solenoid and allows APU compressor discharge air to push the BAV open which illuminates the BLEED VAL OPEN light.

The ECU regulates the APU pneumatic output (bleed air) by sensing APU EGT and comparing this temperature to a preset schedule within the ECU. If EGT becomes excessive, the ECU will command the BAV to close to a one-half open position and allow the EGT to cool down. As EGT cools below a preset value for 20 seconds, the ECU will command the BAV to a full open position.

NOTE

After the APU is started and operating (loaded), APU fuel flow will automatically input approximately 110 pph into the UNS-1Csp FMS system.

Airplane Engine Starts — APU Generator On Line

On Ground Starts

The APU generator may be used as a source of electrical power to start the airplane engines. Starting the first engine on the ground with the APU generator online, is an APU generator start only. The battery disconnect relay ”opens” when the engine start is initiated to prevent the battery from becoming involved during the start (Figure 6-21).

Starting the second engine with both the APU generator and the first engine generator on line, will involve both generators only assisting the engine start. Again, the battery disconnect relay opens as the second engine start is initiated (Figure 6-22).

Initiating engine starts on the ground with the APU generator on line, will cause the APU start relay to close (APU RELAY ENGAGED light illuminates) to provide electrical power for engine starts via the battery bus and the engine start relay(s). The start logic is the same as an

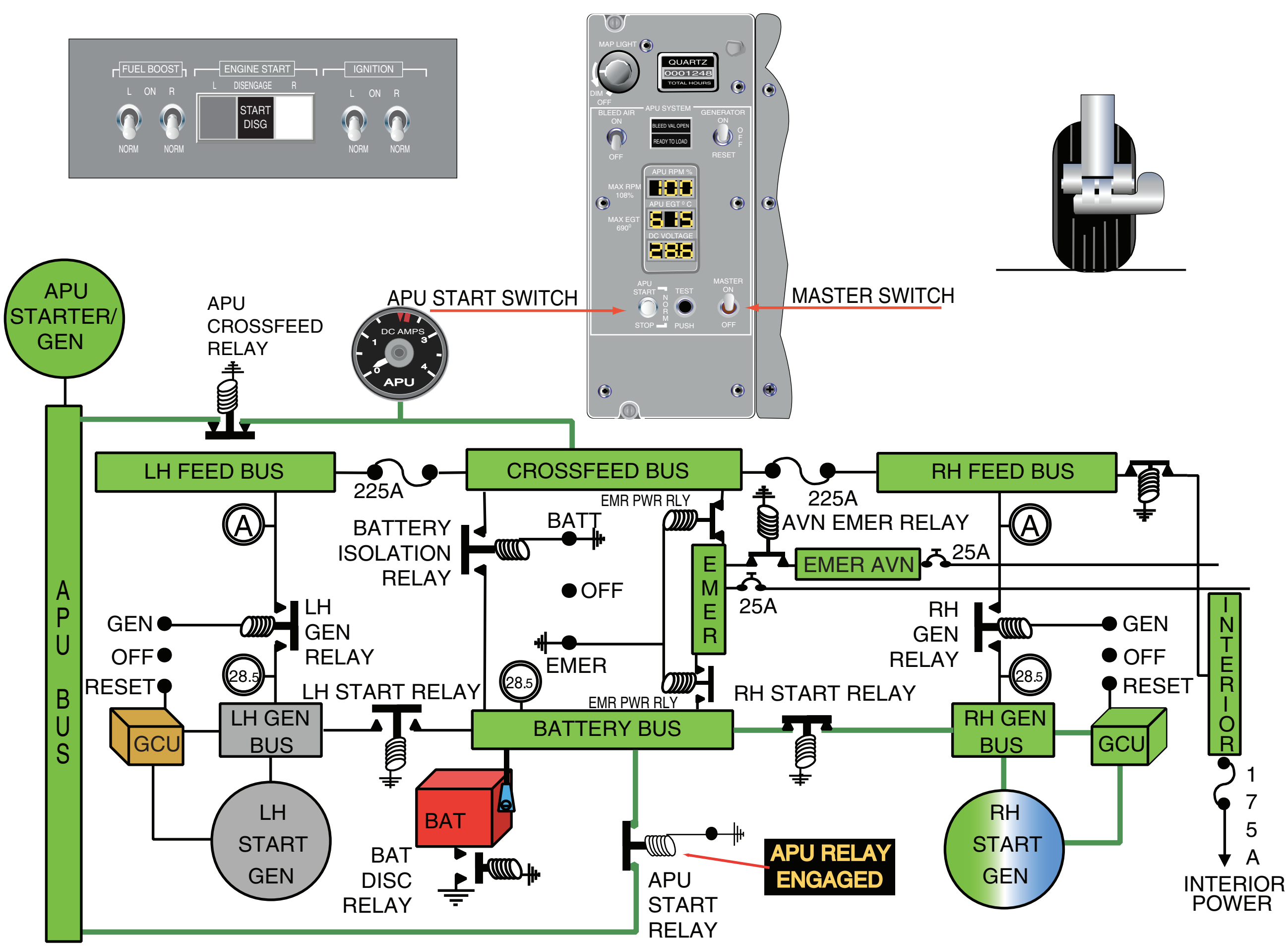


Figure 6-21 APU Gen On Line; Right Engine Start — On Ground

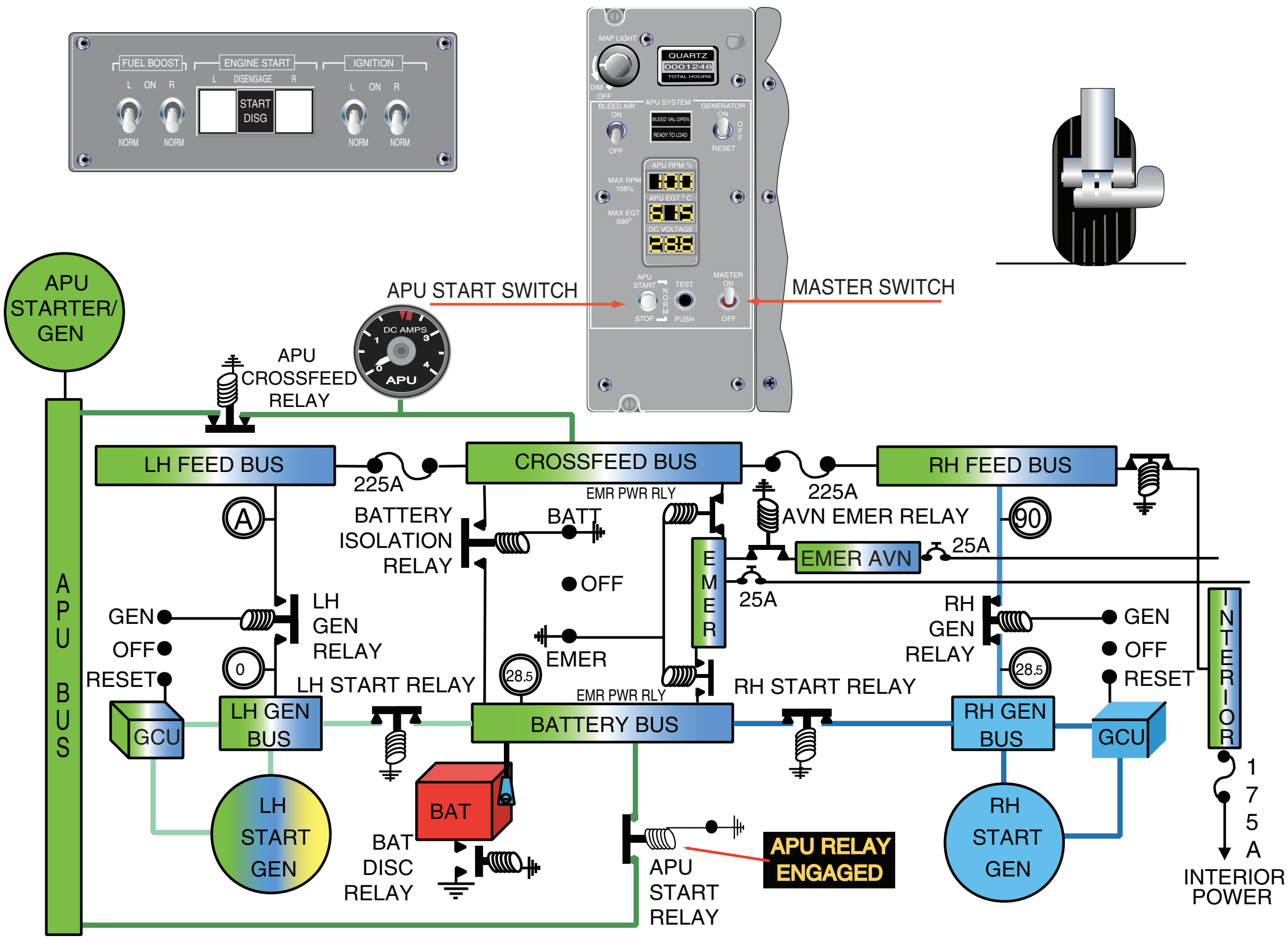


Figure 6-22 APU Gen On Line; Left Engine Start — On Ground (RH Gen On Line)

engine generator assisted start, except the battery is not involved (APU RELAY ENGAGED and engine start button(s) will illuminate).

In Flight Starts

APU starts in flight, and engine starts in flight with or without the APU generator on line, are strictly battery starts (squat switch logic). Starting the APU and or engines in flight, squat switch logic prevents the operating engine(s) or APU start relay(s) from closing thus preventing the operating engine(s) or APU from assisting the battery during starts. This logic is required in order to maintain adequate generator power on the main DC bus system (Figure 16-20).

NOTE

An APU start attempt in flight is prohibited after a dual generator failure.

APU STOP SEQUENCE

APU stop logic and sequencing is identical whether or not the APU is shutdown on the ground or in flight.

Prior to shutdown, the APU should be unloaded. The APU BLEED AIR switch is selected OFF and the BLEED VAL OPEN light extinguishes. The APU GENERATOR switch is placed OFF to unload the generator. The APU generator may

be verified off line by observing the APU ammeter (no load).

The APU is normally shut down by momentarily placing the APU START–STOP switch to STOP. The ECU commands a shutdown by originating an overspeed test signal which deenergizes the APU fuel solenoid and the fuel torque motor closed. The right hand wing fuel boost pump also trips OFF. Placing the APU START–STOP switch to STOP will extinguish the READY TO LOAD light.

After the APU rpm has rolled down to 0%, the ECU is primed for another command signal. After rolldown is completed, the MASTER switch may be selected OFF.

NOTE

After a commanded shutdown using the APU START–STOP switch, the ECU remains powered until the APU MASTER switch is placed OFF.

NOTE

Following an APU shutdown for any reason, a restart must not be attempted until 30 seconds after the rpm indicator displays 0%.

NORMAL PROCEDURES

PREFLIGHT

During the preflight/exterior inspection, insure the APU engine air and generator cooling air inlets, and the APU exhaust outlet (all located on the rear fuselage above the right engine pylon) are clear. Check that the tailcone ram air inlet below the right engine pylon on the fuselage is clear.

An APU oil level check is conducted using the APU maintenance panel, located just inside the tailcone door, adjacent to the right side of the electrical J-box above the light switch. If during the oil level check, the “amber” ADD OIL light illuminates, the APU may be operated and the oil reservoir serviced at the next available opportunity. If the “red” LOW OIL light illuminates, the APU may not be operated until the oil system is serviced.

NOTE

Inaccurate oil level indication may be observed if not checked within five minutes of shutdown.

OPERATION

The APU may be operated on the ground or in flight. In flight operation is limited to FL300.

APU STARTING

The APU may be started on the ground by battery power only, GPU only, or a combination of aircraft generator(s) assisted battery starts.

NOTE

Any time the aircraft battery is involved in an APU start, either alone or generator(s) assisted, it counts as one-third of a normal engine start against the battery. The battery is limited to nine APU start cycles per hour. A GPU start does not involve the battery.

Cold Weather Starts

Cold weather battery starts have been demonstrated to -30°C. Starts below this temperature will probably drain the battery whereby main engine starts using the battery may not be possible. In which case, a ground power unit (GPU) will be required.

If starting the APU below -40°C, it is recommended that Type I (MIL-L-7808) lubricant be used.

APU Starts (Ground or in Flight)

In flight APU starts are battery only starts. Squat switch logic prevents generator-assisted starts (Figure 6-20).

NOTE

In-flight APU starts are limited to FL200 and below.

NOTE

In-flight APU starts are prohibited after a dual generator failure.

Insure the APU generator and bleed-air switches are both OFF.

Place MASTER switch **ON** and **push** the **TEST button**. A valid test is indicated by illumination of the following lights:

- APU FIRE **(red)**, if fire-detection circuit is valid
- APU RELAY ENGAGED **(amber)**
- APU FAIL **(amber)**
- BLEED VAL OPEN **(white)**
- READY TO LOAD **(white)**

And the control panel digital indicators display:

- **RPM - 49EGT - 500 ± 10; DC VOLTS - 00.0**

To initiate the start process, momentarily place the APU START–STOP switch to START and release. The APU RELAY ENGAGED light will illuminate (APU start relay engaged). The **LO FUEL PRESS–R** annunciator extinguishes, and the APU rpm indicator will begin increasing followed by an APU EGT indication at light off. At 50% rpm, starter cutout speed, the APU RELAY ENGAGED light extinguishes. As the APU rpm increases to 95% plus four seconds (rpm should stabilize at 100%) the READY TO LOAD light illuminates. At this point, the start is complete.

NOTE

If the engine generator(s) are online and operating, the main engine start light(s) will illuminate when starting the APU on the ground (generator assisted battery starts). In flight, only the battery provides APU start power.

After the READY TO LOAD light illuminates, the APU generator and/or the APU bleed air switch(es) may be placed ON.

After the APU generator switch is placed ON, the APU ammeter gage on the copilot's instrument panel should be checked for maximum amperage load (not to exceed 200 amps on the ground or 230 amps in flight).

NOTE

Placing the APU generator switch ON, results in the APU generator relay closing and the generator output connects to the crossfeed bus. If the engine generator(s) are on line, the APU generator will parallel with the engine generator(s).

Placing the APU bleed air switch ON, allows the APU Bleed Air Valve (BAV) to open (indicated by the BLEED VAL OPEN light). APU bleed air is now being supplied to the ACM, temperature control valves, and the service air system.

NOTE

With the APU running, bleed air is supplied to the aircraft service air system regardless of the position of the BLEED AIR switch (ON or OFF).

NOTE

With the engines and the APU running, the PRESS SOURCE selector in NORM, LH or RH, and the APU bleed-air valve open, APU bleed-air pressure will override engine bleed-air pressure during taxi, takeoff and climb until APU bleed-air pressure falls below regulated engine bleed air at an altitude of approximately 18,000 - 20,000 feet.

While operating in the configuration noted above, the APU will supply bleed air to the service air system during low throttle settings at low altitudes.

NOTE

APU operation is limited to FL300 and below.

During high humidity when cool air is required, fog may form at each wemac outlet. Selecting a warmer temperature should reduce the fog.

Due to high air flow rates from the APU, selecting BLEED AIR ON may cause a small pressure bump in the cabin. The pressure bump may be minimized by first selecting a temperature colder than present cabin temperature, or manual full cold in extreme cases. Also, closing the cabin door while the APU is operating with the bleed valve open will cause pressure bumps in the cabin. Opening the pilot(s) side window(s) prior to closing the cabin door and then slowly closing the window(s) will minimize pressure bumps.

APU Shut Down (Ground or in Flight)

Prior to shutdown, the APU is unloaded by placing the APU BLEED AIR and GENERATOR switches OFF. The BLEED VAL OPEN light should extinguish and the APU ammeter should indicate no load. Insure the APU RELAY ENGAGED light is extinguished.

The APU START–STOP switch is momentarily placed to STOP.

CAUTION

AFTER MOMENTARILY PLACING THE SWITCH TO STOP, POSITION THE SWITCH TO OFF RATHER THAN ALLOWING IT TO SPRING BACK TO THE OFF POSITION. THIS PROCEDURE SHOULD PREVENT THE SWITCH FROM MOVING PAST “OFF” AND MOMENTARILY STRIKING THE START POSITION, CAUSING THE APU TO RESTART.

A stop command is initiated by the ECU that simulates an overspeed signal that deenergizes the APU fuel solenoid and the fuel torque motor. The READY TO LOAD light extinguishes.

After the APU rpm rolls down to 0%, the APU MASTER switch may be placed OFF.

NOTE

Following shutdown for any reason, an APU restart must not be attempted until 30 seconds after the rpm indicator reads 0%.

EMERGENCY/ ABNORMAL PROCEDURES

EMERGENCY PROCEDURES

APU Fire (APU FIRE WARNING Illuminated)

1. APU Fire Warning Switch — LIFT cover and PUSH

WARNING

THE AIRPLANE BATTERY MUST BE INSTALLED AND THE BATTERY SWITCH IN BATT POSITION OR THE AIRPLANE GENERATOR(S) MUST BE ON AND OPERATING PRIOR TO AND DURING ALL APU OPERATION TO ASSURE FIRE PROTECTION SYSTEM POWER.

NOTE

The APU FAIL light will illuminate when the APU FIRE BOTTLE is discharged.

2. APU — VERIFY SHUTDOWN (RPM 0%, EGT decreasing).
3. APU Bleed Air Valve Switch — OFF
4. APU Generator Switch — OFF
5. APU Master Switch — ON
6. If APU Fire Warning Light Remains Illuminated — LAND as soon as possible.
7. APU Master Switch — OFF (after landing).

Battery Overheat (BATT O'TEMP Light On)

NOTE

Initiate procedures below after basic BATTERY OVERHEAT procedures have been complied with in the *AFM* Section III or the *Pilot's Abbreviated Checklist* (tab M1).

1. Electrical Load — REDUCE
2. APU Generator — OFF
3. APU Bleed Air Switch — OFF
4. APU START/STOP Switch — STOP
5. APU RELAY ENGAGED Light — VERIFY OFF
6. Battery Switch — OFF

ABNORMAL PROCEDURES

APU FAIL Light Illuminated Prior to APU Start

Indicates either fire bottle pressure low or ECU detected a fault.

1. Do not attempt APU start until condition is corrected.

APU FAIL Light Illuminated While APU Was Running

Indicates that the APU automatically shutdown.

1. APU — VERIFY SHUTDOWN (RPM 0%, EGT decreasing).
2. APU Bleed Air Switch — OFF
3. APU Generator Switch — OFF

APU Fails to Shut Down With APU STOP Switch (Ready to Load Light Remains Illuminated)

1. APU MASTER Switch — OFF
2. ADVISORY — Indicates the overspeed sensor has failed. Do not restart APU until condition is corrected.

APU Relay Engaged Annunciator Remains Illuminated After Start On Ground

1. Main Engine Generators — OFF (or Shutdown Engines)
2. Verify GPU Disconnected
3. Battery Disconnect Switch (LH Panel) — LIFT GUARD AND DISCONNECT

NOTE

At this point, all ship's electrical power should be off and the APU should shut down.

If APU Continues to Run

4. APU START/STOP Switch — STOP POSITION (momentarily)
5. APU — VERIFY APU SHUTDOWN (RPM 0% and EGT Decreasing)
6. APU MASTER SWITCH — OFF
7. MANUALLY DISCONNECT BATTERY — PRIOR TO BATTERY SWITCH, OFF
8. Battery Switch — OFF.

If APU Shuts Down

4. MANUALLY DISCONNECT BATTERY — PRIOR TO BATTERY SWITCH, OFF
5. BATTERY SWITCH — OFF

In Flight

If APU Relay Engaged (annunciator does not extinguish)

WARNING

DO NOT SHUTDOWN APU UNTIL ON GROUND.

After Landing

1. PERFORM ON GROUND CHECKLIST PROCEDURES

APU Bleed-Air Valve Open (BLEED VAL OPEN Light Illuminated

1. ADVISORY — Indicates the APU bleed-air valve is in the OPEN position.

Prior to APU Start

2. APU Bleed-Air Switch — OFF

LIMITATIONS

GENERAL

1. APU operation is prohibited until a satisfactory APU test has been accomplished as contained in the NORMAL PROCEDURES section of SUPPLEMENT 16 of the *Excel Airplane Flight Manual (AFM)*.
2. Starting the APU is prohibited whenever the APU FAIL light is illuminated.
3. APU start attempt is prohibited after a dual generator failure.
4. Following shutdown for any reason, APU restart must not be attempted until 30 seconds after the rpm indicator reads 0%.
5. Applying deice (anti-ice fluid of any type) is prohibited with the APU operating.

6. Deployment of the thrust reversers for more than 30 seconds with the APU operating is prohibited.

7. The APU is not approved for unattended operation.

8. The following limits apply to APU starting and operation:

APU OPERATING LIMITS

See Table 6-1.

BATTERY AND APU STARTER CYCLE LIMITATIONS

Starter Limitation

Three APU start cycles per 30 minutes. Three cycles of operation with 90-second rest periods between start cycles is permitted.

Battery Limitation

Nine APU start cycles per hour. (An APU battery start counts as 1/3 of a normal engine battery start.)

NOTES

1. On the ground, no battery cycle is counted when starting the main engines using a cross generator start from the APU generator or from a ground power unit.

2. Use of an external power source with voltage in excess of 28 VDC or current in excess of 1000 amps may damage the starter. Minimum 800 amps for start.

3. If battery limitation is exceeded, a deep cycle including a capacity check must be accomplished to detect possible cell damage. Refer to Chapter 24 of the *Excel Maintenance Manual* for procedure.

Table 6-1 APU OPERATING LIMITS

OPERATING CONDITION	MAX ALT FT	MAX EGT°C (NOTE 3)	N_1%	FUEL TEMP °C	MAX GEN LOAD AMPS (NOTE 2)	AMBIENT TEMP °C
STARTING	20,000	690	---	REFER TO BASIC *AFM* FUEL LIMITS	---	-54 TO 54
RUNNING	30,000	690	108	REFER TO BASIC *AFM* FUEL LIMITS	200 GRD (NOTE 1) 230 FLT	-54 TO 54

NOTES

1. Transient current greater than 200 amperes is approved for APU cross generator start of the main engines.
2. APU Ammeter Instrument Markings:

   a Red Triangle = 200 amperes

   b.Red Line = 230 amperes
3. APU will automatically shut down if EGT limits are exceeded.

CHAPTER 7
POWER PLANT

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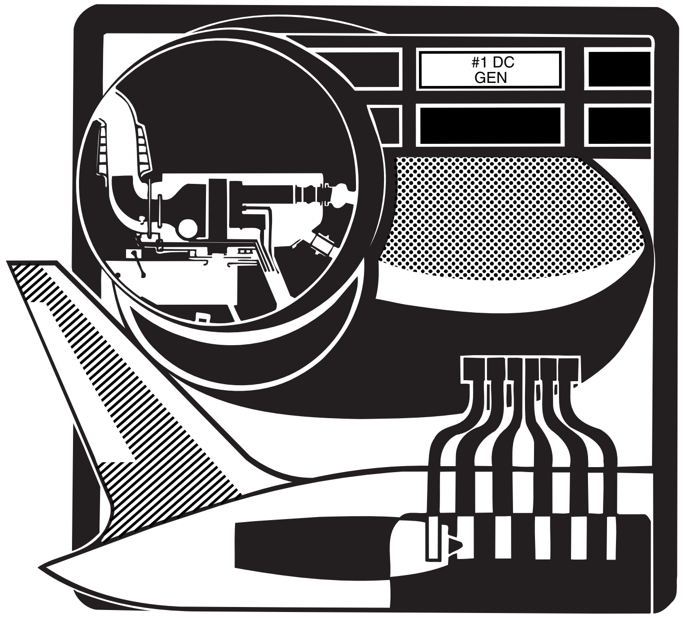
CHAPTER 7
POWERPLANT



INTRODUCTION

This chapter deals with the Citation EXCEL powerplants. It includes information regarding powerplant systems such as engine oil, fuel and ignition, monitoring, engine power control, starting, and engine synchronization.

GENERAL

The Citation EXCEL aircraft utilizes two Pratt & Whitney Aircraft of Canada Inc., PW545A engines for propulsion. The engines are lightweight, nine-stage, twin-spool turbofans that develop 3,804 pounds of flat-rated thrust at sea level to 83°F (28°C). The PW545A engine TBO is currently established as 5,000 hrs with Hot Section Inspection (HSI) at 2,500 hours.

ENGINE DESCRIPTION

The PW545A is a twin-spool, low-noise, high-bypass ratio, turbofan engine that incorporates a full length annular bypass duct. The low-speed fan section is driven by a three-stage turbine assembly, and the high-speed compressor section is driven by a single-stage high-pressure turbine. Engine power is normally controlled through an Electronic Engine Control (EEC) device with

provisions for manual control directly through the hydromechanical Fuel Control Unit (FCU). The engine incorporates a reverse flow annular combustion chamber to reduce weight. The turbine exhaust assembly incorporates a lobe-style forced mixer which enhances performance. This mixer assembly reduces turbulence by gradually mixing hot core air and bypass flow for a smoother exhaust evacuation and noise level reduction.

The PW545A engine is assembled using computerized part stacking procedures for the two rotor assemblies which optimizes rotor alignment and minimizes rotor plane deviations and significantly reduces vibration levels.

ENGINE SPECIFICATIONS:

- Maximum dry weight—830 lbs (376.5 kg)
- Takeoff rated thrust—3804 lbs, Sea Level to 83°F (28°C) — test cell rating
- Maximum continuous rated thrust — 3,767 lbs, sea level to 83°F (20°C)—uninstalled rating
- Bypass ratio — 4.0 : 1
- Oil quantity (system) — 6.13 qts (5.8L)
- Oil quantity (tank) — 2.44 qts (2.3L)

GENERAL OPERATION

The twin-spool design incorporates two major rotating assemblies that comprise the heart of the engine. One assembly consists of the two-stage low-pressure (LP) compressor consisting of a single stage fan and an axial booster stage driven by three low-pressure turbines, commonly referred to as the N_1 section. The other assembly consists of the three-stage high pressure (HP) compressor consisting of two axial compressor stages and one centrifugal stage compressor driven by a single-stage high-pressure turbine wheel, commonly referred to as the N_2 section. The two rotor assemblies are not mechanically connected. The low speed N_1 rotor assembly shaft that connects the low pressure turbines to the fan section, travels through the hollow center core of the HP rotor assembly shaft. This concentric shaft arrangement allows for a free wheeling N_1 rotor assembly. The two rotating assemblies rotate at different speeds and in opposite directions

The intermediate case contains an integral accessory gearbox driven by the high-speed rotor assembly. All engine-driven accessories are mounted on the accessory gearbox. They include the engine oil pumps and engine fuel pump with an integral associated Fuel Control Unit (FCU). Also included on each engine accessory gear box are the starter/generator, alternator, and hydraulic pump.

The PW545A engine is normally controlled by a single channel electronic engine control (EEC) that regulates low rotor (fan) speed in response to pilot controlled Throttle Lever Angle (TLA). However, engine rpm can be fully controlled in the reversionary (manual) mode by the hydromechanical Fuel Control Unit (FCU). The FCU governs high rotor (HP) speed and schedules fuel flow directly from mechanical throttle inputs.

An integral oil tank located in the intermediate case, provides lubrication to the bearings and gears.

Air entering the engine is accelerated rearward by the fan and discharged through two passages. One passage directs airflow through the outer passage (bypass flow) where it flows rearward and is directed through a full length annular bypass duct to generate thrust (Figures 7-1 and 7-2).

Air flow entering the inner passage (core) passes through a booster, through two axial flow compressors, and then to the centrifugal high pressure compressor. High pressure air is then discharged into the annular reverse flow combustion chamber through 22 diffuser pipes. A compressor bleed valve is incorporated in the intermediate case to prevent compressor surge by discharging air into the bypass duct. A supply of core air is used for engine cooling and sealing.

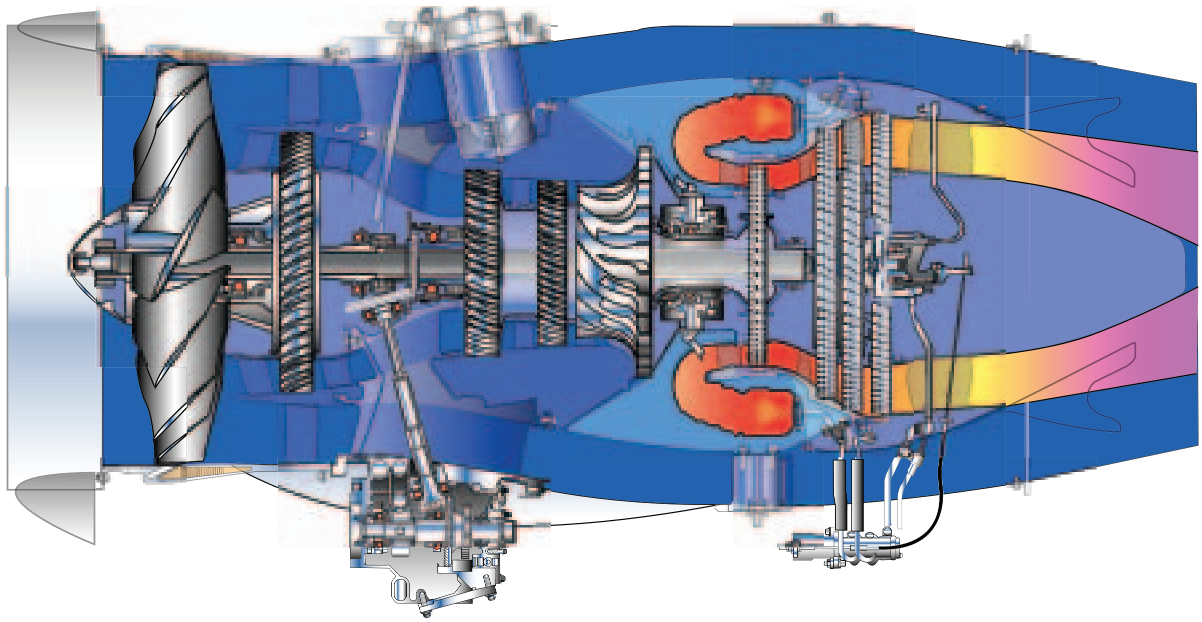


Figure 7-1 PW545A Cross Section

In the combustion chamber, core air is mixed with fuel by 11 hybrid fuel nozzles. The mixture is ignited by two spark igniters which protrude into the combustion chamber. The combustion gases expand and accelerate through the high pressure turbine guide vane ring which directs high velocity gases to provide rotational energy on the high pressure turbine. The high pressure turbine in turn drives the three-stage high-pressure compressor assembly. The expanding gases accelerate rearward through the low pressure guide vanes and rotate the three low pressure turbines to drive the fan and booster. The hot gases are then directed to atmosphere through the exhaust forced (lobe) mixer. The hot gases from the core mix with compressed air from the bypass duct to provide total thrust of the engine (Figure 7-1).

Approximately 60% of thrust produced at sea level is from bypass air and 40% from core airflow through the exhaust port. At 45,000 feet, approximately 40% of thrust is produced through the bypass duct and 60% from core air through exhaust.

The engine is started by activating the starter on the accessory gear box to spin the HP rotor. As rpm increases, ignition and fuel are introduced to complete the combustion process to drive the turbines and, through the concentric shaft arrangement, spin the compressors. When the high-speed compressor reaches sufficient speed to sustain engine operation, the starter and ignition source are automatically switched off. At that point, the engine produces thrust as requested from the cockpit by throttle movement. The EEC governs LP rotor speed by modulating fuel flow via the FCU to the combustion chamber. Engine shutdown is accomplished by cutting fuel off to the combustion chamber from the cockpit (throttles to cutoff).

ENGINE SECTIONS

COMPRESSOR SECTIONS

The engine compressor section (cold section) consists of the low pressure (LP) rotor assembly, intermediate case, LP shaft with Nos. 1, 2, and 5 bearings high pressure (HP) compressor, and the gas generator case.

Low Pressure (LP) Compressor

The function of the LP compressor is to supply air flow to the core and bypass sections of the engine to produce thrust. The fan is an integral bladed (19-fan blades) rotor machined from a solid block of titanium. Attached is an aluminum nose cone that is anti-iced continually during engine operation. The fan is meticulously balanced and fan tip clearance can be increased or decreased with an adjusting spacer located between the fan shaft and the hub. A booster stage consisting of 55 booster blades is mounted on the LP shaft directly behind the fan to increase air pressure into the core. The fan stages are supported by the Nos. 1 and 2 bearings.

Low Pressure (LP) Rotor Assembly (N_1)

The LP rotor assembly consisting of the inlet cone, LP compressor, LP booster, LP shaft, low pressure turbines (3), and LP bearings and seals is detail balanced during engine assembly. During the first test cell run, any remaining unbalance is corrected by adding counterweights to the fan front balancing flange. These weights are then recorded on the engine data plate and in the engine log book. The fan can be replaced in the field without the need of a balance check run. The LP assembly rotates counter clockwise (looking forward) driven by the three low pressure turbines connected by the LP shaft and supported by the Nos. 1, 2, and 5 bearings.

Fan Case

The abraidable sound-proofing fan case permits shrouding of the LP compressor, establishes tip clearance specifications, and provides containment in case of blade fracture.

There are stainless steel inner guide vanes attached that directs primary air flow through the booster toward the HP compressor inlet. The first and second set of inner guide vanes are anti-iced by HP compressor discharge (P_3) air selected by the pilot (see Chapter 10, ICE AND RAIN PROTECTION). There are a set of aluminum outer guide vanes that direct air flow toward the bypass duct. The outer guide vanes are not anti-iced.

Intermediate Case

The intermediate case is the main structural case of the engine and machined from magnesium casting. The case supports the Nos. 1, 2, and 3 bearings, fan case, gas generator case and the bypass duct. The accessory gear box is integrally casted as part of the bottom section of the intermediate case and a "saddle" type oil tank is incorporated. A mount pad is provided for the $P_{2.8}$ compressor Bleed Off Valve (BOV) assembly and a bracket located at Top Dead Center (TDC) is provided to mount the engine vibration sensor. The case has four front-mounted engine attachments to install the engine on the airframe (Figures 7-2 and 7-3).

High Pressure (HP) Compressor Assembly (N_2)

The HP compressor's function is to increase core air pressure and direct it to the gas generator for combustion. HP gas provides sealing of bearing cavities, hot section cooling, high-pressure bleed air for engine and airframe anti-icing/deicing, and air for cabin bleed systems (P_3). The HP compressor section consists of two axial flow compressors coupled to a centrifugal compressor. The three stage HP compressors are driven by a single stage HP turbine connected directly by the HP shaft (Figure 7-1). The HP compressor assembly is supported by the Nos. 3 and 4 bearings. The HP rotor assembly rotates clockwise, opposite of the LP rotor.

Compressor Bleed Off Valve (BOV)

The compressor Bleed Off Valve (BOV), mounted on the intermediate case, allows for surge-free operation of the engine throughout all operating conditions. The BOV is pneumatically operated but normally controlled electrically by the EEC. The EEC controls the BOV position based on corrected N_2 speed and ambient conditions. It is designed to bleed $P_{2.8}$ air (secondary air from the inlet side of the HP compressor into the bypass duct) (Figure 7-1).

The EEC commands the torque motor to regulate P_X air (controlled air through the torque

motor) to the BOV (Figure 7-4). P_X air is regulated P_3 air (HP compressor discharge air).

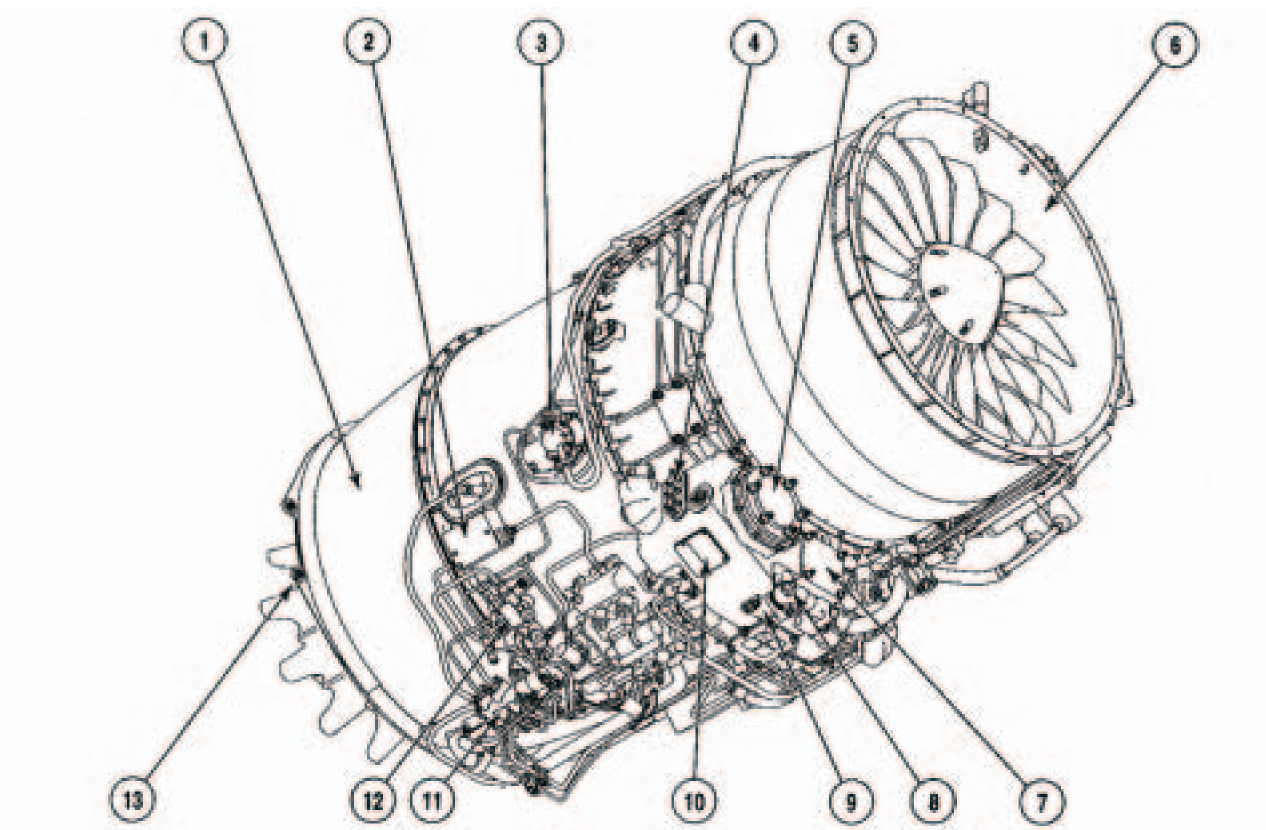
In the pneumatic back-up mode (EECs off line), the EEC signal to the BOV is lost and the torque motor assumes a fixed neutral position. P_X pressure is now directly proportional to P_3 air pressure. P_X pressure is sufficient to keep the BOV closed. In the event of a compressor surge, P_3 decreases and $P_{2.8}$ air increases until it exceeds P_X air and forces the BOV open until the surge condition disappears.

NOTE

If the BOV sticks open, there will be a corresponding increase of approximately 30° increase in ITT on the affected engine.

NOTE

If the BOV sticks closed, it will be very difficult to prevent compressor stalls, "surges," when power is reduced.



1. REAR BYPASS DUCT
2. FUEL FLOWMETER
3. ENGINE P_3 AIR SERVICES
4. OIL LEVEL SIGHT GLASS
5. ALTERNATOR DRIVE PAD
6. FAN CASE
7. STARTER/GENERATOR DRIVE PAD
8. CHIP COLLECTOR
9. OIL TANK DRAIN
10. DATA PLATE
11. EMERGENCY FUEL SHUT-OFF VALVE
12. FDV-FUEL FLOW DIVIDER
13. REAR ENGINE MOUNT

Figure 7-2 PW545A Right Front View

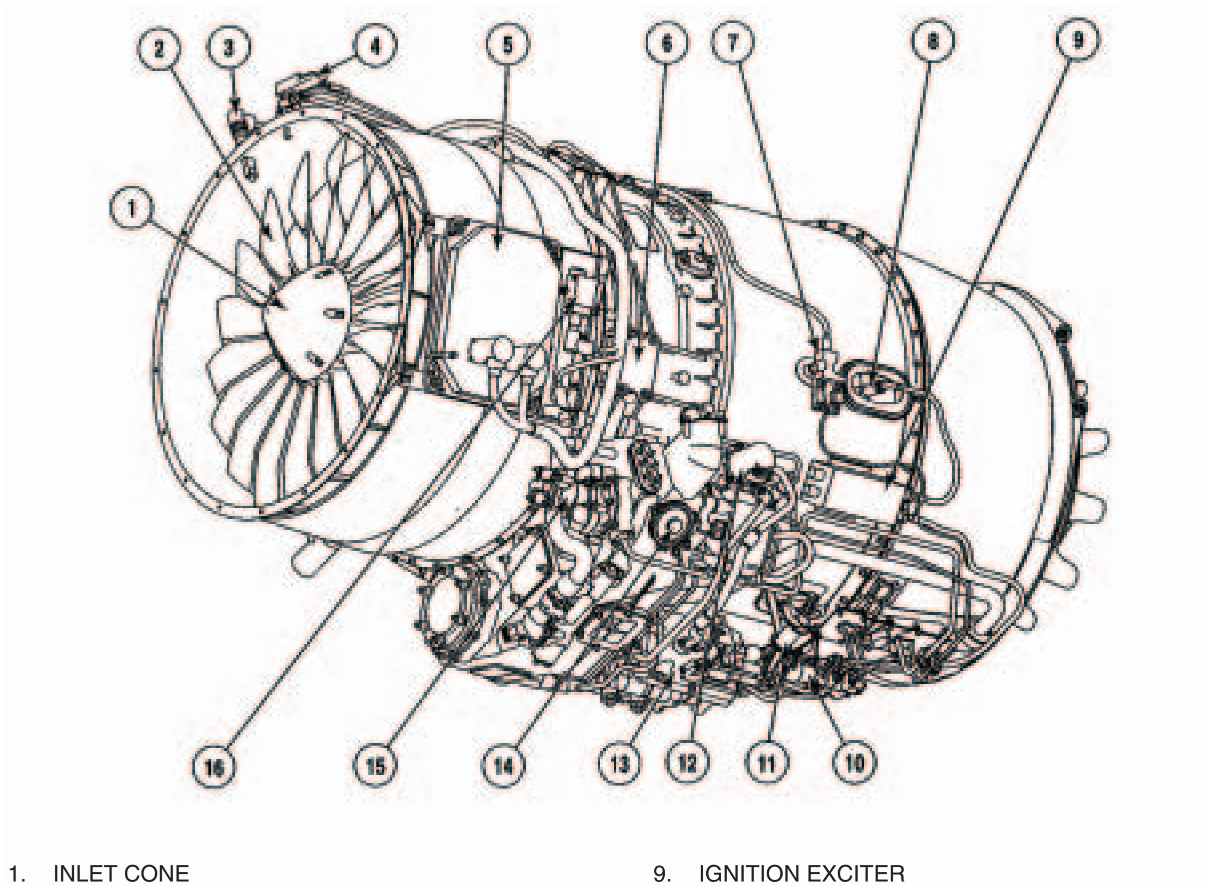
Gas Generator Case

The gas generator case contains P_3 air supply from the HP compressor through the 22 diffuser ducts. The case also houses and supports the combustion chamber, turbine case assembly, and the igniter support tubes.

COMBUSTION AND TURBINE SECTIONS

Combustion Section

The combustion section (hot section) is contained in the rear of the gas generator case. The hot section consists of a combustion liner, a turbine case assembly and HP turbine vane guides.



1. INLET CONE
2. LOW PRESSURE COMPRESSOR
3. T.O. SENSOR
4. T_1 THERMOCOUPLE SENSOR
5. ELECTRONIC ENGINE CONTROL (EEC)
6. ENGINE MOUNT PAD
7. T_1/T_6 THERMOCOUPLE SENSOR
8. IGNITER PLUG
9. IGNITION EXCITER
10. EMERGENCY FUEL SHUTOFF VALVE
11. POWER LEVER LINKAGE
12. FUEL FILTER HOUSING
13. FUEL CONTROL UNIT (FCU)
14. OIL FILTER COVER
15. ENGINE OIL PUMPS
16. DATA COLLECTION UNIT (DCU)

Figure 7-3 PW545A Left Front View

The combustion chamber liner is constructed of nickel alloy and designed in a reverse flow annular configuration. A ceramic coated thermal barrier is applied to the inside of both the inner and outer liners. The ignition plugs pass through support tubes at the 4 and 8 o'clock positions. eleven hybrid airblast fuel nozzles protrude into the combustion chamber liner. The turbine case houses the hot section components which include the fuel nozzles, fuel manifold and the HP turbine liner. It also provides support for the turbine exhaust duct assembly (Figure 7-5).

The HP turbine vane ring is integrally casted from nickel alloy and contains air cooled guide vanes to accelerate and direct combustion gases to the HP turbine.

Turbine Section

The turbine section consists of a single-stage HP turbine disk rotor (first stage), a three-stage LP turbine rotor (second, third, and fourth stages), and an exhaust assembly.

The HP turbine disk is designed to pull energy from the combustion chamber expanding gases to turn the HP compressor assembly and the Accessory Gear Box (AGB). The HP disk assembly is single stage uncooled with 70 directionally solidified nickel cobalt alloy disks. It is connected to the HP impeller by external splines. It is detail balanced during engine assembly with classified counterweights and recorded in the engine log book. It is replaceable in the field without the need for a balance check run. It rotates clockwise.

The LP turbine rotor assembly pulls energy from the HP turbine exhaust gases to turn the LP compressor assembly (fan assembly). It consists of three LP turbine disk rotors with interstage guide vanes supported between the LP turbines. One vane guide ring is installed upstream of the LP rotors to direct expanding gas flow from the HP turbine to the first stage LP rotor disk assembly.

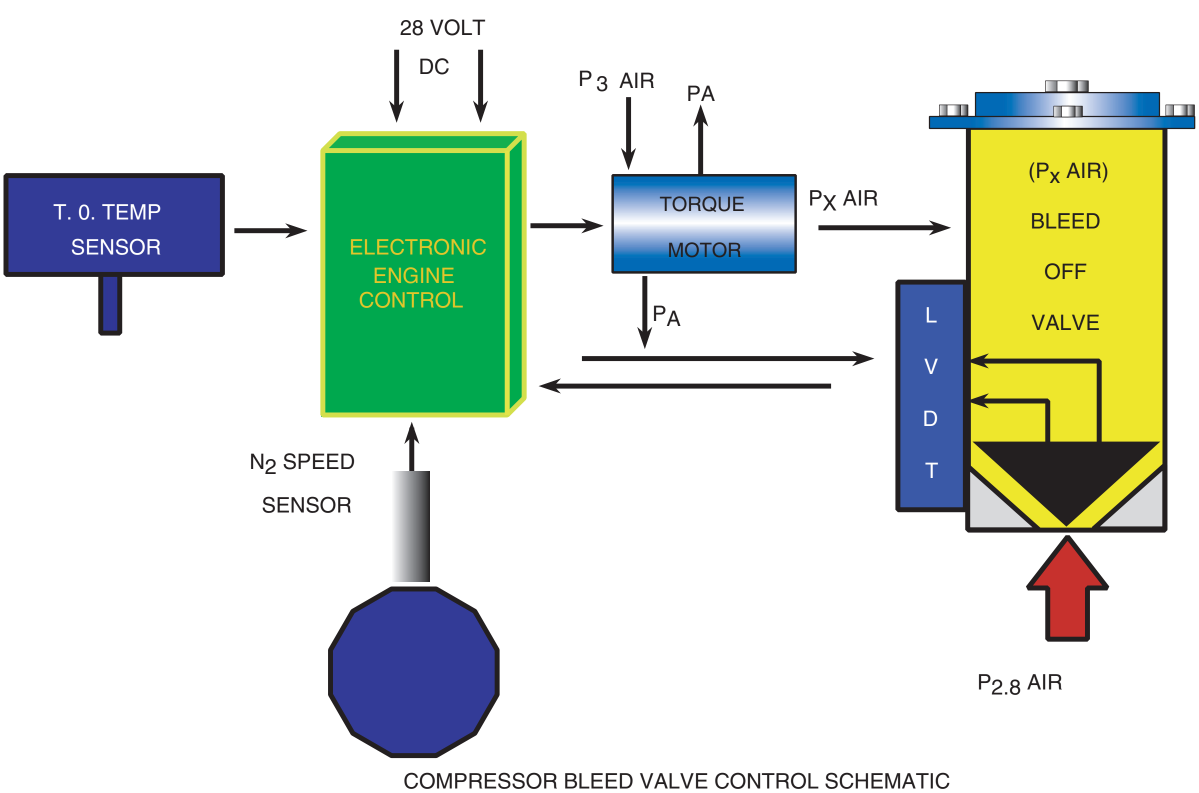


Figure 7-4 Compressor Bleed Valve Schematic

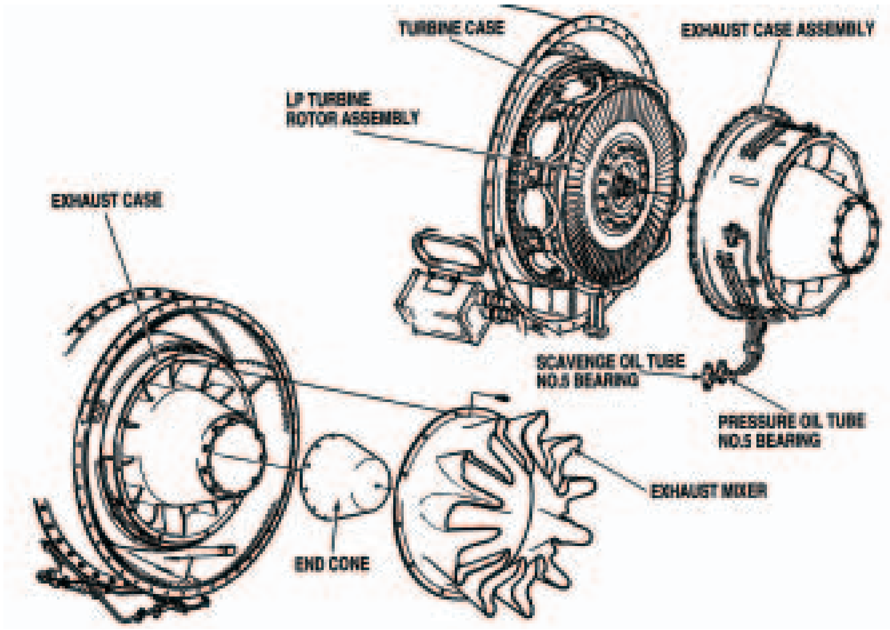


Figure 7-5 Turbine Exhaust Assembly

Turbine Exhaust Assembly

The turbine exhaust assembly consists of an exhaust case, and exhaust "lobe" mixer, and an exhaust cone (Figure 7-5). The assembly is comprised of an inner and outer wall joined by hollow struts. The 6 o'clock strut provides passage for the No. 5 bearing oil scavenge tube and the mechanical shut-off cable (Figure 7-6). The 9 o'clock strut provides passage for the No. 5 bearing oil pressure tube. The case also supports the T_6 thermocouples (supplies exhaust temperature sensors for ITT computation). The forced "lobe" exhaust mixer gradually mixes turbine exhaust "core" air with bypass air flow for increased engine performance.

TOWER SHAFT AND ACCESSORY GEARBOX

Tower Shaft

The N_2 tower shaft meshes with the HP rotor and the Accessory Gearbox (AGB) at the starter/generator gear, all within the intermediate case. Both ends of the tower shaft are equipped with bevel gears and joined by a coupling shaft via a spline at both ends to connect to the HP rotor and the AGB. The tower shaft serves the purpose of turning the HP rotor during start, and upon completion of the start, allows the HP rotor to drive the AGB.

Accessory Gearbox (AGB)

The Accessory Gearbox (AGB) is an integral part of the intermediate case and is equipped with various gears, seals, bearings and drive shafts. It is lubricated by the engine oil system as detailed below. The AGB supplies drive pads for the following accessories:

- Fuel control and integral fuel pump
- Oil pumps (1 pressure and 4 scavenge)
- Hydraulic pump
- Impeller breather (oil/air separator)
- Starter/Generator
- Alternator

The AGB also supplies installation pads for:

- Fuel/Oil Heat Exchanger—(FOHE)
- Chip collector

Breather air is discharged rearward through a breather tube into the engine exhaust.

ENGINE SYSTEMS

Engine systems include the following:

- Fuel System
- Power Control
- Lubricating (Oil) System
- Secondary Air System
- Ignition System
- Indicating Systems (Monitoring)
- Thrust Reversing (Chapter 13, Hydraulic Power Systems)

ENGINE FUEL SYSTEM

The main components of the fuel system are the engine-driven fuel pump, hydromechanical Fuel Control Unit (FCU), Flow Divider Valve (FDV), fuel manifold, fuel nozzles, and the emergency fuel shut-off mechanism.

Engine-Driven Fuel Pump

The engine-driven fuel pump is a two-stage pump integral with the FCU mounted on the AGB. The fuel pump (low pressure stage) receives fuel supply from the wing fuel tank at low pressure and delivers high-pressure fuel from the high-pressure stage to the FCU (Figure 7-7). The pump is not a suction pump and must receive fuel under pressure to operate. In the unlikely event the pump fails, the engine will flame out.

Between the low-pressure stage and the high-pressure stage is a Fuel/Oil Heat Exchanger (FOHE) that receives fuel and allows the warmer oil to heat the fuel. The warmed fuel then passes through a 10-micron fuel filter and onto the high pressure stage of the pump which delivers fuel to FCU (Figure 7-7).

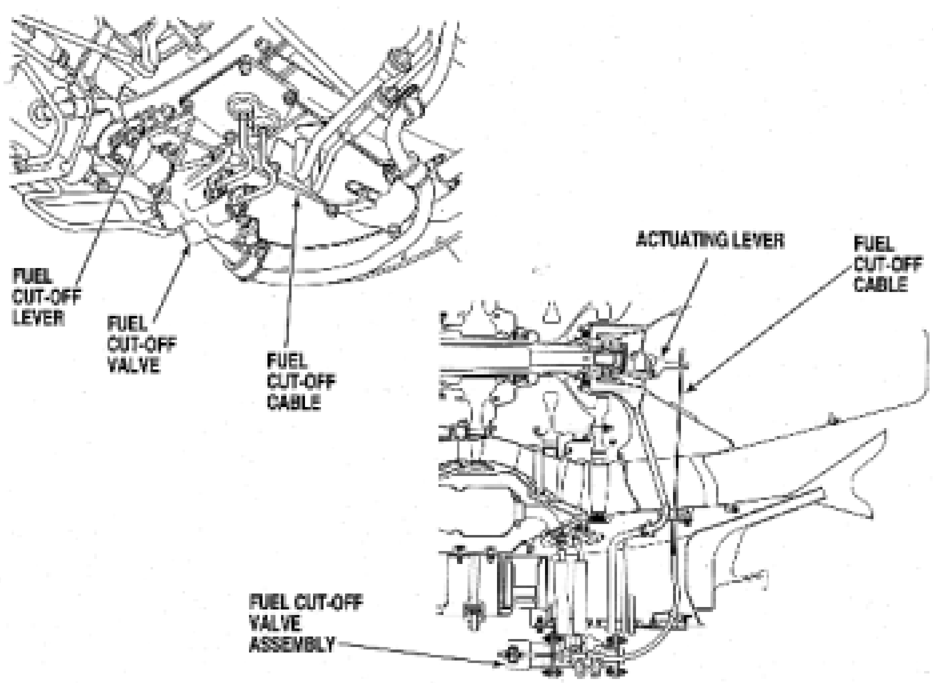


Figure 7-6 Emergency Fuel Shut Off

NOTE

If an engine is shutdown in flight and windmilling longer than 15 minutes without the wing fuel boost pump ON and supplying fuel to the engine pump, the engine-driven fuel pump must be inspected after landing. The fuel filter between the low-pressure stage and the high-pressure stage of the fuel pump is equipped with a bypass feature. If a restriction occurs in the filter, the **FUEL FLTR BP L** or **R** annunciator will illuminate to notify the pilot that a serious situation is developing.

Fuel Control Unit (FCU)

The hydromechanical FCU receives high pressure fuel from the fuel pump and delivers fuel to the flow divider. It interfaces with the fuel/oil heat exchanger and motive flow fuel supply to the wing tank (refer to Chapter 5, FUEL SYSTEM). The flow divider valve splits metered fuel flow between the primary and secondary manifolds that direct fuel to the fuel nozzles. It also acts as a dump valve to drain the manifolds during engine shutdown (Figure 7-7). An ambient remote T1 FCU temperature sensor is flush mounted in the engine nacelle lip and provides temperature input to the FCU in case of EEC or T.O. failure (Figure 7-3).

Flow Divider Valve

The flow divider valve regulates fuel flow from the FCU to the primary and secondary manifolds. The flow divider valve utilizes P_3 air to regulate fuel to the respective manifolds. At the beginning of engine start, fuel is supplied to the primary manifold only. As the start progresses, P_3 air pressure is increased, and the flow divider begins supplying fuel to the secondary manifold, consequently both manifolds are charged at approximately 26-28% N_2.

Engines are shutdown by pulling the throttle back to cutoff. A drain valve is incorporated into the flow divider to drain residual manifold fuel

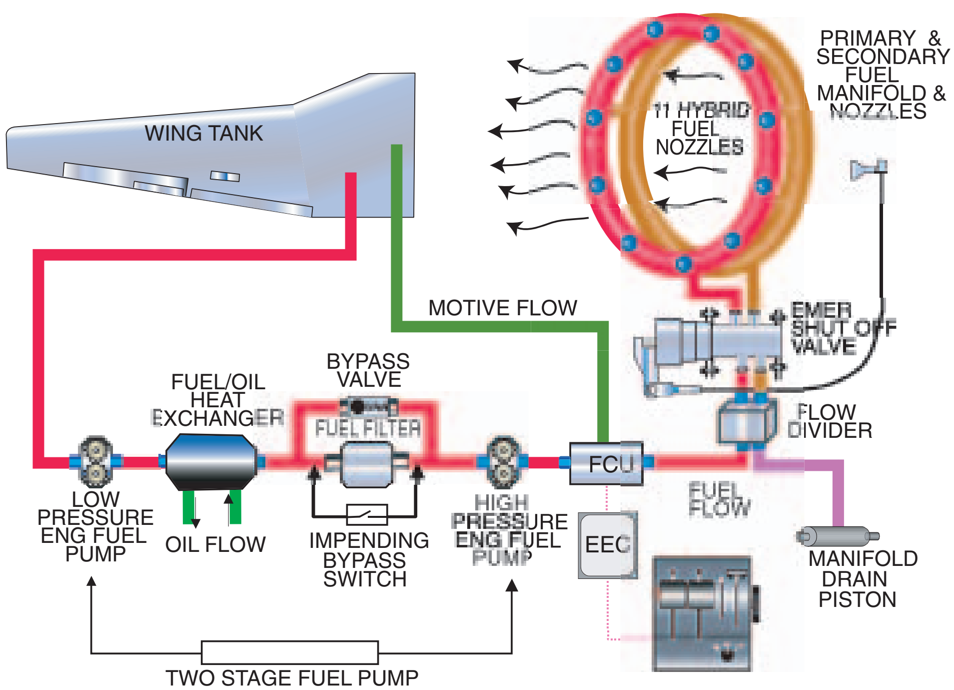


Figure 7-7 Engine Fuel System Schematic

into a fuel holding reservoir when the engine is shut down. During engine starts, the residual stored fuel is directed back to the manifold.

Fuel Manifold

The fuel manifolds, primary and secondary, deliver high pressure fuel to the 11 hybrid fuel nozzles to atomize fuel within the combustion chamber. Fuel delivered to the nozzles is under extreme pressure in order to mix with the high air pressure being injected into the combustion chamber. Primary fuel through the primary port of the fuel nozzles is atomized utilizing fuel pressure. Secondary fuel atomization relies on the flow of P_3 air through the combustion chamber liner.

Emergency Shutoff Valve

Primary and secondary fuel passes through a normally open shutoff valve from the flow divider prior to reaching the fuel manifold. The shutoff valve is operated mechanically by aft displacement of the low-pressure turbine shaft (0.07 inches). It may shift aft if a major malfunction occurs such as, overspeed or decoupling of the low pressure turbines from the LP shaft.

A plunger is installed through the No. 5 bearing housing and it borders the extremity of the low pressure (N_1) turbine shaft. A cable is attached to the rear of the plunger and passes through the 6 o'clock exhaust strut to the mechanical fuel "emergency" shutoff valve (Figure 7-6). Rearward displacement of the low pressure turbine shaft will strike the plunger and activate it to a "tripped" position thus drawing the emergency shutoff valve closed. The shutoff will remain in the tripped (closed) position until reset by maintenance personnel.

Fuel Flow Indication

A flow meter senses metered fuel flow downstream of the FCU and displays fuel flow in pounds per hour on a dual LCD gage on the center instrument panel (Figure 7-8).

Power source for the fuel flow gauges is supplied by main DC power with circuit breaker protection located on the pilot's circuit breaker panel. A loss of main DC power will cause the gauges to go blank.

Figure 7-8 Engine Instruments

NOTE

Fuel flow indication is disabled when the associated throttle is moved to cutoff. This prevents erratic fuel flow indication as the rpm decreases below 10%.

ENGINE CONTROL SYSTEM

Electronic Engine Control (EEC)

The function of the engine control system is to control the engine low rotor (N_1) speed throughout various throttle positions and prevailing ambient conditions. Engine power control is a function of the Electronic Engine Control (EEC) system (Figure 7-9). The EEC is a single channel, digital microprocessor based controller that provides two modes of operation: AUTOMATIC (AUTO) and MANUAL (MAN). MANUAL mode may be pilot selected by the ENGINE COMPUTER switch located on the pilot's lower instrument panel (Figure 7-10). If a major fault in the EEC system occurs, the system will automatically revert to manual (MAN) mode. Manual mode is annunciated by a "white" EEC MANUAL L/R advisory light in the annunciator panel.

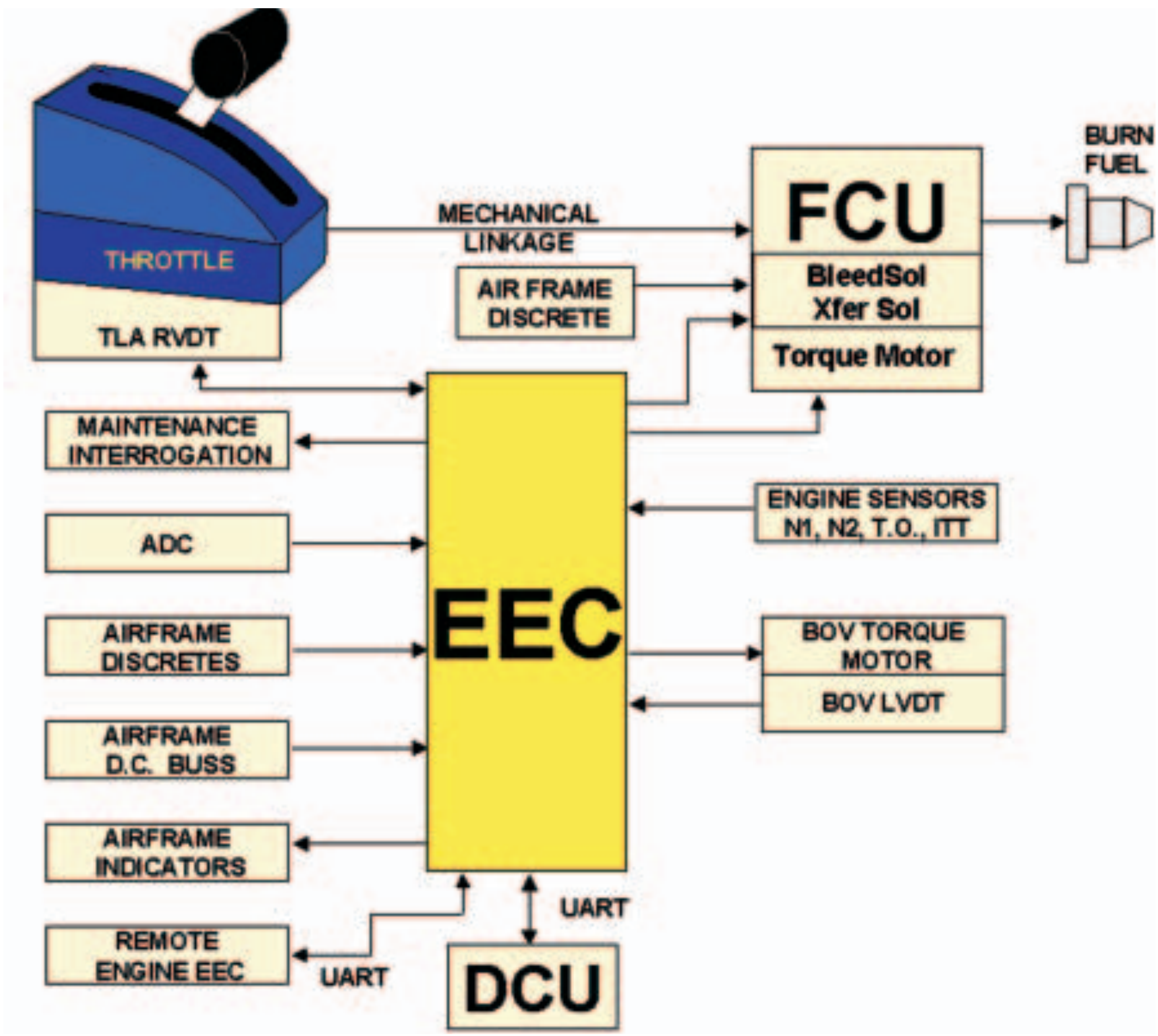


Figure 7-9 Control System Schematic

In AUTO mode, the EEC provides the following functions in response to throttle movement (Thrust Lever Angle—TLA).

1. Detented throttle positions, automatic thrust settings (N_2 governing), engine anti-ice system ON or OFF.
2. Automatic idle governing (N_2 at ground idle, flight idle, and anti-ice idle).
3. Acceleration and deceleration limiting.
4. N_1 and N_2 limiting.
5. Closed loop bleed valve (BOV) control.
6. Engine diagnostic system (EDS) functions.
7. Overspeed protection (N_2).
8. N_1 or N_2 synchronization.

The following inputs are monitored and processed by the EEC:

1. Fan inlet temperature - T.O.
2. Power lever angle position (throttle) - TLA
3. ITT
4. N_1 speed
5. ADC data (Pressure Altitude, ambient Temperature, Mach No.)
6. Discrete inputs (bleed air state, squat switch, sync and Thrust Reversers)

Throttle Detents

In AUTO mode, TLA is provided electronically to the EEC by a Rotary Variable Displacement Transducer (RVDT) located below the throttle quadrant. The engine mounted EEC schedules fuel flow by referencing N_1.

Throttle detent pneumatic air cylinders, use regulated service bleed-air pressure to position detent plates in the throttle quadrant to establish

Figure 7-10 Pilot's Switch Panel

three throttle position detents: CRUISE, CLIMB and TAKE-OFF. The EECS will maintain Maximum Takeoff Thrust (N_1), Maximum Continuous Climb Thrust (N_1), or Maximum Continuous Cruise Thrust (N_1) while the throttle is in the applicable detented position regardless of altitude, ambient temperature, airspeed, and anti-ice ON or OFF. The EEC receives air data information from the airplane air data computers (ADC 1 hardwired to L EEC and ADC 2 hardwired to R ECC).

NOTE

The pilot should insure the EECs are providing proper N_1 power settings by periodically comparing N_1 chart data to the N_1 gages while operating in AUTO with the throttles set in the detents.

A shutoff valve will close if the EEC reverts to MANUAL mode and shutoff bleed air to the throttle detent actuator. When bleed air is removed from the actuator, spring action pulls the detent plates out of the way. *If maximum power is required while operating in MANUAL mode, the appropriate thrust charts will need to be consulted in the AFM or Pilot's Abbreviated Checklist.*

CAUTION

WHEN OPERATING IN MANUAL MODE, THROTTLE WILL BE ABNORMALLY SENSITIVE IN THE HIGH-POWER RANGE.

Ground Idle

The EEC controls low speed ground idle in AUTO mode which automatically decelerates the engine to an idle speed of approximately 47% N_2 (sea level) while on the ground, and the throttle reduced to the idle stop (left squat switch, Ground-On-Ground — GOG).

A "white" advisory **GND IDLE** light is illuminated in the annunciator panel during low-speed ground idle operation (EECs AUTO mode). Upon landing, there is an eight-second delay before the engines decelerate to low speed ground idle (LH squat switch). If the EEC switch is repositioned from MAN to AUTO with the aircraft on the ground, the eight-second delay also occurs before the engines decelerate to low-speed ground idle. Low-speed ground idle provides lower taxi speed, especially during colder temperatures.

NOTE

Operating on the ground in MANUAL will result in flight idle rpm on the ground (**GRD IDLE** annunciator extinguished).

NOTE

During take-off (AUTO mode), the **GRD IDLE** annunciator will remain illuminated until airborne.

Engine Synchronizer

The EECs (AUTO) control engine synchronizing in or out of the detents in order to match the right (slave) engine to the left (master) engine, by controlling either the N_1 fan or N_2 turbine rpms. The synchronizer continually monitors the engine speeds and adjusts slave engine speed as required. The range capability of the sync actuator is a 4.75% speed range. Engine rpms should be adjusted within 4% if operating out of the detents prior to engaging SYNC.

A rotary FAN-OFF-TURB switch on the center pedestal below the throttle quadrant, actuates the engine sync system. An amber advisory light adjacent to the switch illuminates when the system is activated. OFF drives the sync actuator to the center of its range before shutting the system down. A turbine out of sync is more noticeable in the cockpit. Conversely, a fan out of sync is more noticeable in the cabin, especially the rear seating area.

The engines are evenly matched with both EECs in AUTO and both throttles in the same detent

(TO, CLB, or CRU). The sync switch should be OFF for major power changes and single engine operation.

NOTE

The engine sync switch is required to be OFF during takeoff and landing. The sync system is inoperative with the EEC(s) in MANUAL mode.

Engine Diagnostic System—EDS

The Engine Diagnostic System's (EDS) primary components consist of the engine-mounted EEC and the Data Collection Unit (DCU). The EEC is the primary element of the EDS system and interfaces with the DCU (Figure 7-9). The DCU records and stores fault codes and engine data from the EEC via a UART cable (Figure 7-9). The following diagnostic and health monitoring capabilities are provided:

- Faults annunciation and recording
- Exceedance recording
- Engine condition trend monitoring data
- Engine life cycle recording

Engine data stored in the DCU can be downloaded to a laptop computer from serial ports located on the engine or in the cockpit (center pedestal) to assist maintenance personnel in troubleshooting, recording engine performance trends, and logging engine life cycle data.

ECS Fault Indicators

Electronic Control System (ECS) fault indicators are located just inside and forward of the tail cone access door inside the tail cone. The fault indicators will alert maintenance personnel of the need to download data from the DCU if stored faults and exceedance data is close to exceeding DTU storage limits (80% full). The DCU has the capability of storing approximately 150 hours of combinations of coded faults (events), exceedance parameters and life-cycle data. The DCU is normally downloaded after each 150 flight hours to coincide with airframe inspection schedules.

An ENG DATA SCAN button located on the center pedestal or the center instrument panel, is provided to allow the pilot to insure flight crew noted engine faults are recorded in the DCU. Momentarily depressing the ENG DATA SCAN button will record an engine data "trace" from the EEC for a 40-second period (20-seconds prior and 20-seconds after the button is pressed). Current engine fault and exceedance data will override stored data if the DCU memory is full.

Manual Mode

Major EEC faults are annunciated in the cockpit by "white" advisory **EEC MANUAL L/R** lights in the annunciator panel indicating the EEC tripped to the MANUAL mode. The Fuel Control Unit (FCU) has full authority of engine speeds in MANUAL. The throttle directly controls the FCU by direct mechanical linkage. The FCU controls engine thrust by governing N_2 speeds.

NOTE

Engine sychronization and low-speed ground idle functions are lost and the throttle detents are no longer in place during MANUAL operation.

The FCU applies the following functions in the MANUAL mode:

- Throttle adjustable power settings (no throttle detents), and N_2 governing.
- Idle governing (N_2) at flight idle, on ground or in flight.
- Acceleration and deceleration limiting by speed of throttle movement (ratio unit control).
- N_2 speed limiting.
- Closed loop bleed valve (BOV) control, pneumatic mode.
- Limited engine diagnostic system functions (EDS).

Throttle Lever Inputs

The throttles are manually operated to select engine power settings. Motion of the throttle levers in the throttle quadrant is transmitted to the Rotary Variable Displacement Transformers (RVDT). The RVDTs transmit throttle position to the respective EECs. Throttle movement is also transmitted mechanically to the respective Engine Fuel Control Units (FCU).

Components located at the bellcrank assembly are: throttle switch bank modules, RVDTs, throttle detent pneumatic air cylinders and throttle detent bleed-air shutoff valves (referred as throttle detents earlier).

Two sets of four switches are used for throttle switches. Each switch bank is connected to the bellcrank assemblies and the single common shaft rotates as their respective throttle position to the following systems:

- Pressurization
- Landing Gear
- Speedbrakes
- No Takeoff
- Ignition
- Boost Pump

ENGINE LUBRICATING (OIL) SYSTEM

The engine oil system provides lubrication, cooling and cleaning of all engine bearings and gears. The system incorporates a pressure pump, scavenge pumps, and a secondary air system (Figure 7-11).

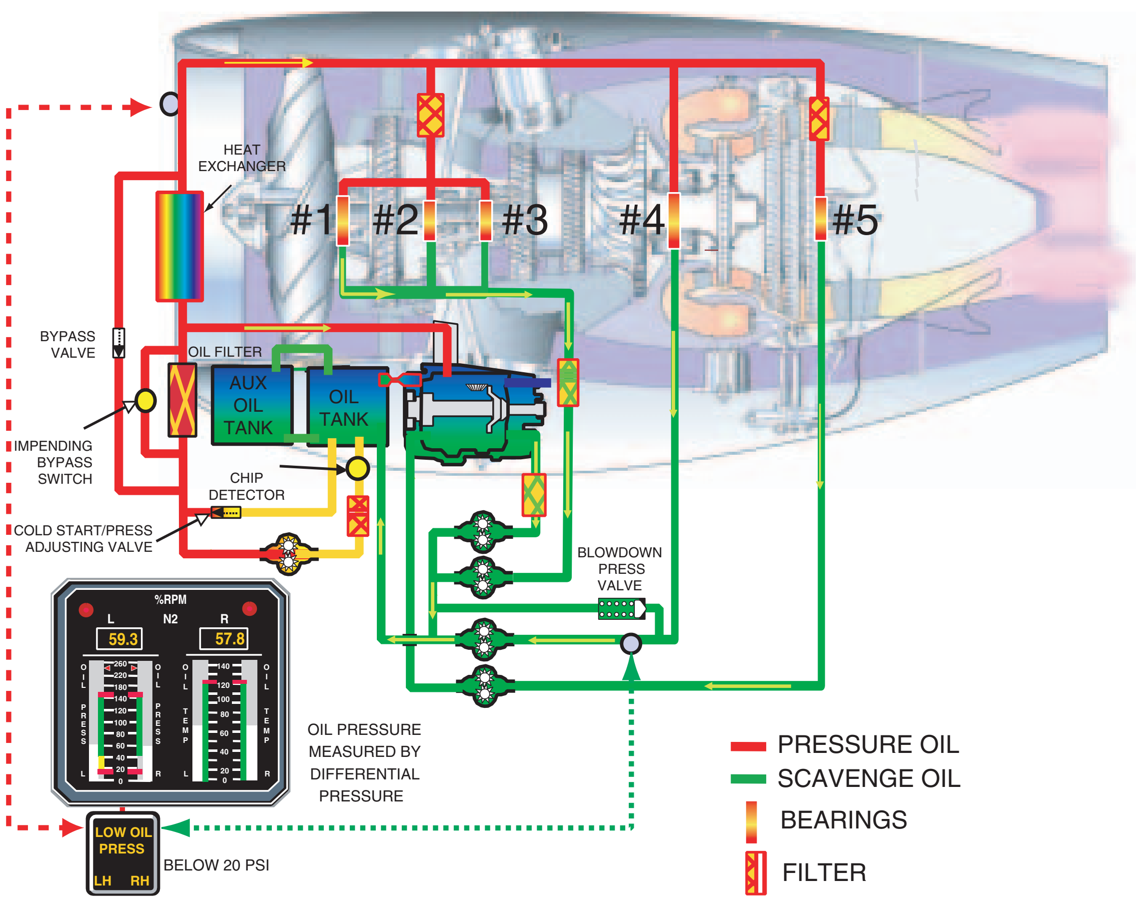


Figure 7-11 Engine Oil Lubricating System

Oil Tank

The oil tank is an integral part of the intermediate case and comprises a saddle-back dual-tank design. There is one tank on each side of the intermediate case and interconnected. Both tanks are equipped with sight indicators to determine quantity (Figure 7-12). This arrangement allows the engines to be installed on either side of the fuselage; however, the nacelle oil access door is only mounted on the outboard nacelle. The engine is canted slightly during installation on the airframe, therefore the outboard tank sight indicator is used to determine accurate oil quantity indications.

Oil Pumps

There are five geared oil pumps (Figure 7-11):

1. Pressure pump
2. Accessory gearbox (AGB) scavenge pump
3. Nos. 1, 2, and 3 bearing scavenge pump
4. No. 4 bearing scavenge pump
5. No. 5 bearing scavenge pump

The pressure pump is flow regulated and supplies oil to satisfy the lubricating requirements throughout the engine operating range. Oil supplied from the oil tank passes through a magnetic chip collector (no associated warning light) prior to reaching the pressure pump. From the pressure pump, oil flows through the main oil filter and fuel/oil heat exchanger (FOHE) before distribution to the bearings.

Calibrated oil nozzles deliver the necessary oil quantity to the various bearings, gears and splines.

Scavenge pumps scavenge oil from the bearings and return the oil directly to the oil tank, by a combination of dedicated scavenge pumps, or indirectly from the No. 5 bearing to the AGB and then to the oil tank from the AGB scavenge pump (Figure 7-11).

Oil Cooling

Oil temperature is maintained within limits by the Fuel Oil Heat Exchanger (FOHE).

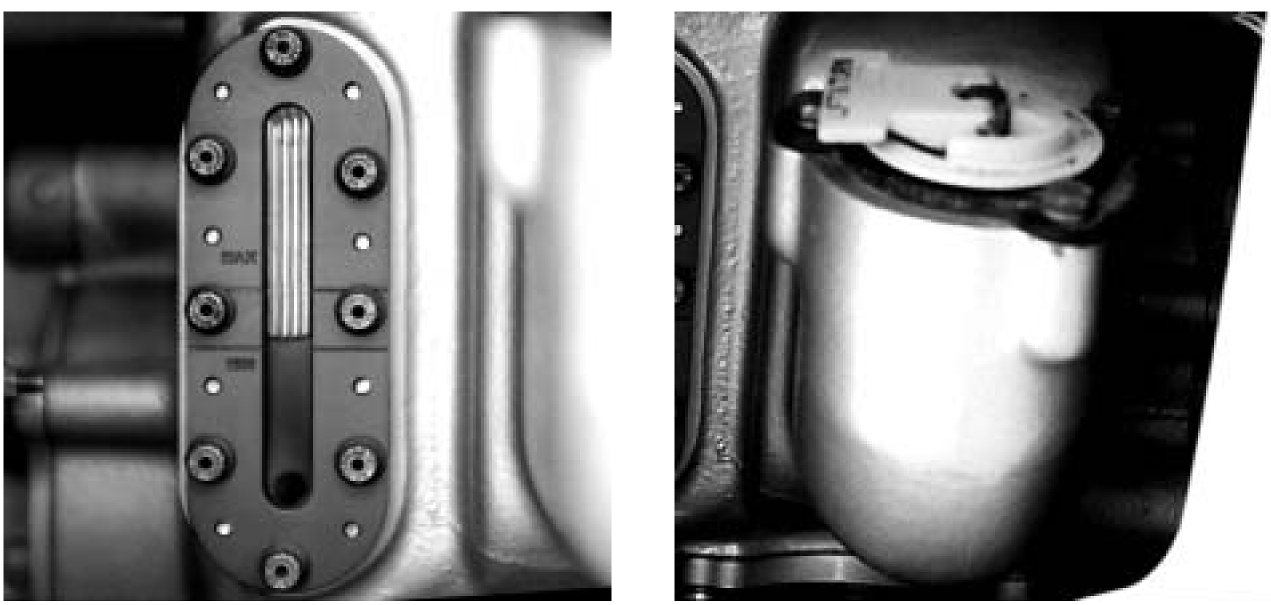


Figure 7-12 Oil Filter and Sight Gauge

Oil Filter

The main oil filter is used to remove solid contaminates from the oil. The filter incorporates an impending bypass switch and bypass valve. The impending bypass switch will activate if the filter begins to clog and trigger the amber **OIL FLTR BP L** and/or **R** annunciator caution light. If the filter becomes blocked to the point that the bypass valve activates, oil may bypass both the filter and the fuel/oil heat exchanger.

NOTE

If the **OIL FLTR BP L** or **R** light illuminates, the crew should monitor oil pressure and oil temperature, and be alert for possible fuel icing. Consider the possibility of partial or total loss of thrust on the affected engine(s).

Oil System Indicators

Oil pressure is sensed by dual transmitters that send signals to the oil pressure indicators located on the center instrument panel (Figure 7-8). The transmitters sense pressure differential from the pressure pump output and scavenge pressure (return) from the No. 4 bearing (Figure 7-11). The indicators are analog vertical tape gages and calibrated in psid. The gages are color coded with a red arrow at 250 psid (transient), red line at 140 psid (maximum continuous), red line at 20 psid (minimum), green stripe between 45 - 140 psid (normal operating range above 60% N_2), and a yellow stripe between 20 - 45 psid (minimum operating range below 60% N_2).

A low oil pressure switch is located adjacent to the oil pressure transmitter and senses pressure differential between the pressure side and the suction side of the oil system. The low oil pressure switch is connected to a warning light in the annunciator panel. If differential pressure drops below 20 psid, the red **LO OIL PRESS L** or **R** annunciator will illuminate and trigger the MASTER WARNING RESET lights.

NOTE

It is normal to observe large pressure changes during major power changes due to the pressure transmitters recording differential pressure (pressure output vs. scavenge return pressure).

Oil temperature is sensed in the pressure line past the FOHE. The oil temp sensor provides a signal to the oil temperature indicator located on the center instrument panel. The oil temperature indicators are analog instruments that display vertical white tape indications adjacent to a temperature scale graduated in degrees Celsius. The scale is colored green from 10° to 121° to indicate normal operating range. A red line at 121° indicates maximum temperature allowed.

NOTE

Engine oils approved for use are listed in the LIMITATIONS section of this manual. Normally, different approved brands must not be mixed.

NOTE

The oil level should be checked as a post flight item. For a valid check, the check should be accomplished 10 minutes after engine shutdown. The maximum allowable oil consumption is 0.2 lbs per hour or 1 qt in a 10 hour period.

Oil system specifications:

- Oil consumption, maximum over a 10 hour period: 0.2 lb/hr, provides an engine endurance of 23.6 hours.
- Oil tank capacity at maximum level indication: 6.13 US qts (5.8 L)
- Usable oil capacity: 2.44 US qts (2.3 L)

SECONDARY AIR SYSTEM

The secondary air system supplies compressed air throughout the engine for nonpropulsion purposes and bleed air extracted for use by various aircraft systems.

Engine Bleed Air (Non-Propulsion)

Labyrinth seals are used as restrictors to control secondary airflow distribution and carbon seals are used for oil retention in bearing cavities.

Compressed air is used to cool hot section components:

- P_3 air is used to cool HP turbine section.
- $P_{2.8}$ air is used to cool the LP turbine section.

P_3 air is used as the servo pressure to modulate the BOV position (Figure 7-4).

Air pressure is used to seal bearing cavities and assist in scavenging oil to the AGB sump. Oil mixed with air is returned to the oil tank and then to the AGB by way of a restrictor (sized to provide tank pressure at altitude to avoid cavitation of the pressure pump). Air is separated from the oil by an impeller/separator mounted on the gearbox and vented to engine exhaust.

The LP compressor nose cone is continually anti-iced by $P_{2.8}$ air flowing through the LP shaft to the inside of the inlet cone while the engine is operating.

The T_1 thermocouple sensor is continually heated by P_3 air any time the engine is operating (see ITT Instrumentation system later in this chapter).

The first and second stage LP compressor inlet guide vanes (aft of the fan, and aft of the booster) are heated by P_3 air through hollow cored passages when selected by the pilot. Air is routed through an anti-icing solenoid valve in the intermediate case plenum to the inner stator vanes. The anti-icing solenoid valve is an "open/closed" design which is electrically commanded closed, and deenergized open/pneumatically assisted by P_3 air. If a loss of main DC power occurs, the anti-icing valve will fail open. Refer to Chapter 10, ICE AND RAIN PROTECTION, for a more detailed description of the engine anti-icing system.

Engine Bleed Air (Airframe)

P_3 ports located on either side of the engine allow bleed air extraction (Figure 7-2). Engine bleed air is used for air conditioning, cabin pressurization, airframe anti-icing including the engines, and the service air system.

IGNITION SYSTEM

Two ignitors protrude into the combustion chamber at the 4 o'clock and the 8 o'clock positions (Figure 7-3). They provide the spark necessary for ignition of the fuel/air mixture in the combustion chamber. The ignition system is a single "dual channel" high energy system. The two ignitor plugs are connected by way of two high tension cables to a single exciter box, mounted on the left side of the fan bypass housing (Figure 7-3). The exciter is a noise suppressed, high voltage, capacitor discharge unit that provides a burst mode type ignition that produces 6-7 sparks per second for initial 30 seconds and then one spark per second continuously thereafter. The unit provides 18,000 to 24,000 volts at 6.0 amps during the 30 second burst mode and two amps in the continuous mode.

The system utilizes 10 to 32 volts DC input. The airplane supply of 24 to 28.5 volts is sufficient to power the system. During engine starts, the voltage drops to approximately 10 VDC. The system is capable of continuous operation when the ignition switch is selected ON or SEC (Figure 7-10) or by the engine anti-ice switch selected to WING/ENGINE ON or ENGINE ON positions (Refer to Chapter 10, ICE AND RAIN PROTECTION).

Any time the ignition system is operating (power to the exciter box), green indicator lights located near the top and adjacent to the ITT indicators are illuminated (Figure 7-8).

NOTE

If one ignitor plug fails during engine starts, the engine will start normally and the ignition light will remain illuminated.

Ignition switches — The three-position NORM–ON–SEC ignition switches are located on the pilot's electrical switch panel (Figure 7-10). In the NORM position, ignition is automatic during starts and selecting the wing/engine anti-ice switch to either the WING/ENGINE ON or ENGINE ON positions. Selecting ignition ON or the wing/engine switch to either of the ON positions requires main DC power supplied through the crossfeed bus. Placing the ignition switch to secondary (SEC) powers the ignition system from the emergency bus.

NOTE

When the green ignition lights located adjacent to the ITT indicators are illuminated, they indicate DC power to the exciter box. They do not necessarily indicate the ignitor plugs are firing.

ENGINE INDICATING SYSTEMS

Engine instrumentation is provided by a horizontal row of gages located in easy view in the top section of the center instrument panel (Figure 7-8). From left to right the gages are:

- N_1 or LP fan rpm — displayed in percentage with white vertical tapes and digitally to the nearest tenth. Green band — 20% to 100%. Red lined at 100%.
- ITT — displayed in degrees Celsius by white vertical tapes. Green band — 0° to 720°. Individual red lines — 720°.
- Ignition — green lights adjacent to each ITT indicator near the top of the scale
- N_2 or HP turbine rpm — digital read out only to nearest tenth expressed in percentage of rpm. Red lights adjacent to each digital window "flash" red if rpm increases to 101.2% .
- Oil Pressure — displayed in psid by white vertical tapes. Green band — 45 to 140 psid. Yellow band — 20 to 45 psid.

  Red arrows — 250 psid, red lines at - 140 psid and 20 psid.
- Oil Temperature — displayed in degrees Celsius by white vertical tapes. Green band — 10° to 121°. Red lines — 121°.
- Fuel Flow — displayed digitally in pounds per hour (pph).
- Fuel Quantity — displayed in pounds by white vertical tapes.

NOTE

All engine indicators require main DC electrical power except the N_1 gages, which are powered by the emergency DC bus system. Circuit breakers for engine gages are located on the pilot's CB panel.

Rotor Speed Sensing System

Two speed sensors, one mounted at the 12 o'clock position on the intermediate case (N_1 sensor) and one located on the accessory gearbox (N_2 sensor) provide speed signals to the cockpit indicators and to the EECs. The sensors are dual coils, induction type (electrical) speed sensors.

N_1 sensor reads the speed of the LP shaft and the N_2 sensor reads the speed at the teeth of the FCU drive gearshaft. The N_2 sensor also sends a speed signal to the EEC to compute correct bleed-off valve position.

Interturbine Temperature Sensing System (ITT)

The ITT measuring system is computed from comparing air temperature entering the engine (T_1) with bypass air temperature ($T_{1.4}$) and adding this comparison to turbine exhaust temperature (T_6). ITT is monitored at station 4.5. The $T_{4.5}$ sensing system comprises two independent measuring systems: T_1 and T_6.

T_1 Temperature

When sensing the inlet air stream bypass duct airflow (T_1) and bypass duct airflow ($T_{1.4}$) temperatures of the bypass air flow, and substracting T_1 from $T_{1.4}$; the temperature rise across the fan in the bypass duct is established. The T_1 and $T_{1.4}$ probes contain three thermocouples connected in series. The rise in temperature across the fan is thus multiplied by three to provide the required data. The T_1 probe located in the nacelle lip is heated with bleed air anytime the engine is operating.

T_6 Temperature

Eight thermocouples installed at station 6 (T_6) are connected in parallel to provide an average exhaust temperature. The thermocouples are installed on the exhaust case and protrude through the case into the exhaust gas path. Adding three times the fan rise temperature, ($T_{1.4}$ - T_1) to the exhaust gas temperature (T_6), will provide a simulated readout of interturbine temperature at the outlet from the HP turbine vane ring ($T_{4.5}$). Therefore, a simulated ITT, $T_{4.5}$ readout is computed from the integrated T_1 and T_6 subsystems wired in series. The following mathematical formula applies to ITT:

$$T_{4.5} = 3(T_{1.4} - T_1) + T_6$$

T_1 = Inlet air stream temp obtained from the T_1 probe.

$T_{1.4}$ = Bypass duct air flow temperature taken from station $T_{1.4}$.

T_6 = Average exhaust gas temperature.

Oil Pressure

See Oil System, this Chapter.

Oil Temperature

See Oil System, this Chapter.

Fuel Quantity

See Chapter V, FUEL SYSTEMS.

Fuel Flow

See Fuel System, this Chapter.

VIBRATION DETECTOR

The engine vibration detector system is mounted Top Dead Center (TDC) on the intermediate case of the engine. The system consists of an engine accelerometer connected to an Engine Vibration Monitor Unit (EVMU). The system is designed to detect minute changes in engine frequency induced by slight rpm changes or vibrations. If the vibration monitor detects engine vibration higher than a predetermined normal level, a white advisory ENG VIB L or R light illuminates in the annunciator panel. The white advisory light will illuminate at a vibration level well below a level hazardous to rotor integrity.

NORMAL STARTING PROCEDURES

DESCRIPTION

Engine starting is a semiautomatic function. Once power is applied to the airplane and the starter switch is pressed, power is applied to the starter until a predetermined engine speed is reached at which time power is released from the starter.

The starter portion of the starter-generator operates from electrical power supplied by the airplane battery, a Ground Power Unit (GPU), the opposite engine generator, or the optional APU generator. The starter-generator drive shaft is splined to match the output shaft inside the accessory gear box (AGB). A flexible drive coupling and shear section is incorporated in the starter-generator between the drive spline and the armature to prevent damage to the AGB if a failure occurs. Starter-generator ground cooling is provided by engine bypass air (see Chapter 2, ELECTRICAL POWER SYSTEMS and/or Chapter 6, AUXILIARY POWER UNIT).

The Generator Control Unit (GCU), in conjunction with the start logic printed circuit

board (PCB), controls the start cycle including field weakening for the start mode and discontinuance of the start cycle when the engine has reached a specific speed.

Engine starting system components in addition to the starter-generator and the GCU, (start PCBs, relays and switches) are located in the aft tailcone main power J-box, and on the pilot's lower instrument (switch) panel (Figure 7-10).

NOTE

The Start Logic PCB provides switching and actuation control during starts for the GCU, start relays, battery isolation relay, external power relay, battery disconnect relay and generator power relays.

Ground Power Units (GPU): Current output should not exceed 1,000 amps or be less than 600 amps. Operating above 6,000 feet MSL, output should not be less than 800 amps.

A built-in voltage GPU monitor will trip the external power relay if GPU output voltage exceeds 32 VDC. GPU voltage should be adjusted to a stable voltage of 28 VDC.

The engine starter pad has a torque limitation of 1,600 inch-pounds. The starter shaft has a shear point set to 1,600 inch-pounds to protect the AGB. Adjusting the GPU output not to exceed 1,000 amps and 28 VDC will enable the starter to deliver a torque of less than 50% of the allowable torque.

OPERATION

First Engine Battery Powered Ground Start

Electrical Action

- Generator switches should be placed GEN ON to allow the GCU to close the power relay automatically after the engine start is complete and bring the generator on line.
- Place the battery switch to BATT and insure the battery voltage is at least 24 volts minimum. The battery switch supplies a ground to the start logic PCB. The start logic PCB then provides a ground which energizes the battery isolation relay and provides power to the cross-feed bus, left and right feed buses, and the emergency bus, all in the aft J-box. Power is further supplied through the main distribution system to the left and right circuit-breaker panels.

NOTE

Power has to be available through the L and R START CBs on the pilot's CB panel in order to energize the respective engine start circuits.

- Pressing the left or right starter switch provides a ground to the start logic PCB that in turn inputs a start command to the GCU. The GCU commands the start logic PCB to supply a ground to engage the start relay. The light in the respective start button illuminates when the start relay is closed. The ground fault disable logic prevents inadvertent shutdowns due to improper ground fault signals.
- The start relay is held energized closed by a ground from the start logic PCB.
- The start relay remains closed applying battery power to the starter until the starter-generator speed sensor signal activates the GCU or the starter disengage switch is actuated removing the start logic PCB ground and causing the start relay to open. The speed sensor signal will activate the GCU at approximately 40-45% N_2. A relay installed in the ground fault circuit prevents nuisance tripping of the GCU during starting.
- The respective generator power relay will close and bring the generator on line when the GCU senses the generator output is equal to system voltage (battery voltage) on the respective feed bus (GCU bus sensing circuit).

Pilot Action

- Generator switches — GEN ON
- Insure the EEC switches are in AUTO.
- Place battery switch to BATT, check battery voltage minimum 24 volts.
- Press the start button for the engine to be started (left or right). Notice the respective starter button light illuminates. The instrument panel lighting will begin to fade as the battery voltage decreases to approximately 10-15 volts. Note the respective **FUEL BOOST L** or **R** annunciator illuminates and the associated **LO FUEL PRESS L** or **R** light extinguishes.
- Observe N_2 speed. As speed reaches 8% N_2, advance throttle to idle. Verify green ignition light illuminates (adjacent to the ITT indicator). Engine should ignite within 10 seconds. Observe ITT and N_2 for sign of "light off."
- Observe N_1 rotation by 25% N_2.
- Observe ITT and N_2 increasing. Insure ITT doesn't exceed limits, 720°.

NOTE

If "light off" does not occur within 10 seconds of advancing the throttle or idle is not achieved within 30 seconds or there is no N_1 speed by 25% N_2, abort the start.

A rapid ITT rise during starts is normal, however, if ITT is rapidly approaching 720°, abort start.

Abort the start by placing the throttle to cutoff, insure ignition lights are out, and continue to motor the engine with the starter for 15 seconds to clear fuel from the combustor. Push the starter disengage switch.

Determine the cause and correct before proceeding.

Allow N_2 to reach zero rpm before attempting restart. Typical N_1 rundown time is approximately 350 seconds and N_2 rundown time is 60 seconds.

Observe starter minimum cool time, 90-seconds, before attempting a restart.

CAUTION

IF ENGINE SPEED INCREASES RAPIDLY DURING STARTS OR AT ANY STEADY STATE SETTING WITH NO THROTTLE MOVEMENT, SHUT DOWN ENGINE IMMEDIATELY.

WARNING

IF INTERNAL ENGINE FIRE FOLLOWS SHUTDOWN (VISUAL INDICATIONS OR HIGH ITT), ENGAGE STARTER AND MOTOR ENGINE FOR 15 SECONDS, DISENGAGE STARTER AND IF THE FIRE PERSISTS, PRESS ENGINE FIRE SWITCH TO ARM FIRE EXTINGUISHERS AND FIRE THE EXTINGUISHER(S) INTO THE NACELLE.

- Check respective **EEC MANUAL L** or **R** annunciator extinguishes as the throttles is advanced. At 40-45% N_2, verify starter disengagement (start switchlight OFF, ignition light OFF, and the respective **FUEL BOOST L** or **R** annunciator extinguished).
- Note the respective **GEN OFF L** or **R** annunciator extinguished. The ammeter gage should reflect a load on that side and voltmeter should indicate 28.5 VDC.
- Verify oil pressure is within starting limits.

- Engine rpms should increase and stabilize at approximately 22% ±1.0% N1 and 47.8% ±1.0% N2 with ignition NORM, cockpit bleed air NORM, and anti-icing bleeds OFF. Observe all engine instruments are within limits. Check respective fuel, oil, generator, and hydraulic annunciator lights are extinguished.

Second Engine Cross Generator Assist Start

Electrical Action

CAUTION

THE OPERATING ENGINE MUST BE AT IDLE FOR A CROSS GENERATOR START. TOO HIGH RPM MAY CAUSE GENERATOR OUTPUT POWER TO EXCEED GENERATOR GEAR BOX PAD LIMITS.

- Pressing the start button switch (engine to be started), provides a ground to the start logic PCB that in turn inputs a start command to the GCU. The GCU commands the start logic PCB to supply a ground to engage both start relays. Both start button lights illuminate indicating both left and right start relays are closed. The operating engine generator is now supplying the majority of the power requirements to the opposite starter via the battery bus; however, the battery is still involved in the start. The ground fault disable logic prevents inadvertent shutdowns due to improper ground fault signals.
- The start logic PCB removes the ground from the battery isolation relay and causes it to open to isolate the opposite starter from the main feed bus circuitry and prevents a parallel path of amperage flow through the 225-amp current limiter on the operating side generator feed bus. Due to high amperage loads involved during engine starts, the 225-amp CL would "blow" open if the circuit was not isolated (**AFT J-BOX LMT**).
- The start relays are held energized closed by a ground from the start logic PCB.
- The start relays remain closed applying generator and battery power to the starter until the starter-generator speed sensor signal activates the GCU or the starter disengage switch is pressed removing the start logic PCB ground causing the start relays to open. The speed sensor signal will activate the GCU at approximately 40-45% N_2. A relay installed in the ground fault circuit prevents nuisance tripping of the GCU during starting.
- The respective generator power relay will close and bring the generator on line when the GCU senses the generator output is equal to system voltage (opposite generator voltage) on the respective feed bus (GCU bus sensing circuit).

Pilot Action

- Press the start button for the engine to be started. Notice both starter button lights illuminate. The main feed bus circuitry is isolated from the starter when the battery isolation relay opens, to prevent "blowing" open a 225-amp current limiter.
- Note: the respective **FUEL BOOST L** or **R** annunciator illuminates and the associated **LO FUEL PRESS L** or **R** light extinguishes.
- Observe N_2 speed. As speed reaches 8% N_2, advance throttle to idle. Verify green ignition light illuminates (adjacent to the ITT indicator). Engine should ignite within 10 seconds. Observe ITT and N_2 for sign of "light off."
- Observe ITT and N_2 increasing. Insure ITT doesn't exceed limits, 720°. Observe the same notes, cautions and warnings as stated during the first engine start.
- Observe N_1 rotation by 25% N_2.

- The respective **ECC MANUAL L** or **R** annunciator extinguishes as the throttle is advanced. At 40-45% N_2, verify starter disengagement (both start switchlights OFF, ignition lights OFF, **FUEL BOOST L** and **R** annunciators extinguished).
- Note **GEN OFF L** and **R** annunciators extinguished, the ammeter gages should reflect equal loads indicating the generators are sharing the total load (within 10% of the total) and voltmeter should indicate 28.5 VDC.
- Verify oil pressure is within starting limits.
- Engine rpms should increase and stabilize at approximately 22% ±1.0% N_1 and 47.8% ±1.0% N_2 with ignition NORM, cockpit bleed air NORM, and anti-icing bleeds OFF. Observe all engine instruments are within limits. Check respective fuel, oil, generator, and hydraulic annunciator lights are extinguished.
- Engine annunciators extinguished except the white **GND IDLE** light.

First Engine Ground Power Unit (GPU) Start

Electrical Action

- Insure the GPU is set at 28 VDC and the amperage set knob does not exceed 1,000 amps. Connect the GPU.
- Both generator switches should be OFF to eliminate the generators coming on line after engine start(s) and tripping the GPU off line.
- Placing the battery switch to BATT the voltmeter should indicate 28 volts (insures the external power relay is closed and the GPU is on line). The battery switch supplies a ground to the start logic PCB. The start logic PCB then provides a ground which energizes the battery isolation relay and provides power to the crossfeed bus, left and right feed buses, and the emergency bus, all in the aft J-box. Power is further supplied through the main distribution system to the left and right circuit breaker panels.

NOTE

Power has to be available through the L and R START CBs on the pilot's CB panel in order to energize the respective engine start circuits.

- Pressing the left or right start switch provides a ground to the start logic PCB that in turn inputs a start command to the GCU. The GCU commands the start logic PCB to supply a ground to engage the start relay and cause the battery disconnect relay to open and trip the battery off line. The light in the respective start button illuminates. The ground fault disable logic prevents inadvertent shutdowns due to improper ground fault signals.
- The start relay is held energized closed by a ground from the start logic PCB.
- The start relay remains closed applying GPU power to the starter until the starter-generator speed sensor signal activates the GCU or the starter disengage switch is pressed removing the start logic PCB ground causing the start relay to open and simultaneously closing the battery disconnect relay (battery back on line). The speed sensor signal will activate the GCU at approximately 40-45% N_2. A relay installed in the ground fault circuit prevents nuisance tripping of the GCU during starting.

Pilot Action

- Generator switches — OFF.
- Insure the EEC switches are in AUTO.
- Place the battery switch to BATT, check battery voltmeter 28 volts.
- Press start button for the engine to be started (left or right). Notice the respective starter button light illuminates. Note the respective **FUEL BOOST L** or **R** annunciator illuminates and the as-

sociated **LO FUEL PRESS L** or **R** light extinguishes.

- Observe N_2 speed. As speed reaches 8% N_2, advance throttle to idle. Verify green ignition light illuminates (adjacent to the ITT indicator). Engine should ignite within 10 seconds. Observe ITT and N_2 for sign of “light off.”
- Observe ITT and N2 increasing. Insure ITT doesn’t exceed limits, 720°. Observe the same NOTES, CAUTION and WARNINGS as stated during the first engine battery powered ground start above.
- Observe N_1 rotation by 25% N_2.
- Insure respective **ECC MANUAL L** or **R** annunciator extinguishes as the throttle is advanced. At 40-45% N_2, verify starter disengagement (start switchlight OFF, and the ignition light OFF, respective **FUEL BOOST L** or **R** annunciator extinguished). **GEN OFF L** and **R** annunciators remain illuminated. The voltmeter should indicate 28 VDC and the ammeters should not register.
- Verify oil pressure is within starting limits.
- Engine rpms should increase and stabilize at approximately 22% ± 1.0% N_1 and 47.5% ± 1.0% N_2 with ignition NORM, cockpit bleed air NORM, and anti-icing bleeds OFF. Observe all engine instruments are within limits. Check respective fuel, oil, generator, and hydraulic annunciator lights are extinguished.

Second Engine GPU Start

Electrical Action

- Same action as First Engine EPU Start.

Pilot Action

NOTE

If the generator(s) were in the GEN position, the operating generator will be on line and the GPU will be off line and the second engine start will be a generator assisted cross generator start.

- Press the start button for the engine to be started. Notice only that respective starter button illuminates. The battery isolation relay remains closed. The generator power relays remain open and the GPU is isolated from sending parallel power through the 225-amp current limiters. Note the respective **FUEL BOOST L** or **R** annunciator illuminates and the associated **LO FUEL PRESS L** or **R** light extinguishes.
- Again, observe N_2 speed. As speed reaches 8% N_2, advance throttle to idle. Verify green ignition light illuminates (adjacent to the ITT indicator). Engine should ignite within 10 seconds. Observe ITT and N_2 for sign of “light off.”
- Observe ITT and N_2 increasing. Insure ITT doesn’t exceed limits, 720°. Observe the same NOTES, CAUTIONS and WARNINGS as stated during the first engine start.
- Observe N_1 rotation by 25% N_2.
- Insure respective **ECC MANUAL L** or **R** annunciator extinguishes as the throttle is advanced. At 40-45% N_2, verify starter disengagement (the start switchlight OFF, ignition lights OFF, **FUEL BOOST L** and **R** annunciators extinguish).
- Verify oil pressure is within starting limits.
- Engine rpms should increase and stabilize at approximately 22% ± 1.0% N_1 and 47.8% ± 1.0% N_2 with ignition NORM, cockpit bleed air NORM, and anti-icing bleeds OFF. Observe all engine instruments are within limits. Check respective fuel, oil, and hydraulic annunciator lights are extinguished.
- Engine annunciators extinguished except the white **GND IDLE** and **GEN OFF L** and **R**.
- Signal ground crew to disconnect the GPU. After the GPU is disconnected, check voltmeter for battery voltage, 24-25 volts (verifies battery is connected).
- Place both generator switches to GEN ON. GEN OFF L and R annunciators

extinguish and voltmeter indicates 28.5 VDC and ammeters are displaying equal loads (within 10% of the total load).

In Flight Restart - One Engine, With Starter Assist

Electrical Actions

- Engine configuration: throttle-cutoff, generator switch—GEN ON, Firewall Shutoff—open, and ignition switch—NORM.
- Pressing the starter switch provides a ground to the start logic PCB that in turn inputs a start command to the GCU. The GCU commands the start logic PCB to supply a ground to engage the start relay. The light in the respective start button illuminates whenever the start relay is closed. The ground fault disable logic prevents inadvertent shutdowns due to improper ground fault signals.
- The start logic PCB removes the ground from the battery isolation relay and causes it to open to isolate the opposite generator from the main feed bus circuitry and prevent a parallel path of amperage flow through the 225-amp current limiter on the operating side generator feed bus. Due to high amperage loads involved during engine starts, the 225-amp CL would "blow" open if the circuit was not isolated.
- The start relay is held energized closed by a ground from the start logic PCB.
- The start relay remains closed applying battery power only to the starter. *Squat switch logic prevents the opposite start relay from closing, thus preventing the operating generator from becoming involved in the start and causing a "draw down" on the electrical system*. The respective start relay remains closed until the starter-generator speed sensor signal activates the GCU or the starter disengage switch is pressed, removing the start logic PCB ground and causing the start relay to open. The speed sensor signal will activate the GCU at approximately 40-45% N_2. A relay installed in the ground fault circuit prevents nuisance tripping of the GCU during starting.
- The respective generator power relay will close and bring the generator on line when the GCU senses the generator output is equal to system voltage on the respective feed bus (GCU bus sensing circuit).

Pilot Action

- Insure throttle - CUTOFF, generator switch — ON, firewall shutoff — OPEN, and ignition switch — NORM
- Insure the EEC switches are in AUTO.
- Momentarily press the start button. Notice the starter button light illuminated and the operating engine start switch light remains out. Note the respective **FUEL BOOST L** or **R** annunciator illuminates and the associated **LO FUEL PRESS L** or **R** light extinguishes.
- Observe N_2 speed. As speed reaches 8% N_2, advance throttle to idle. Verify green ignition light illuminates (adjacent to the ITT indicator). Engine should ignite within 10 seconds. Observe ITT and N_2 for sign of "light off."
- Observe ITT and N_2 increasing. Insure ITT doesn't exceed limits, 720°.
- Observe N_1 rotation by 25% N_2.
- Insure respective **ECC MANUAL L** or **R** annunciator extinguishes as the throttle is advanced. At 40-45% N_2, verify starter disengagement (starter switch light OFF, ignition light OFF, and the respective **FUEL BOOST L** or **R** annunciator extinguished).
- Note **GEN OFF L** and **R** annunciators extinguished, the ammeter gages should reflect equal loads indicating the generators are sharing the total load (within 10% of the total) and voltmeter should indicate 28.5 VDC.

NOTE

The generator will come on line automatically if the generator switch is in GEN ON position and the engine was not previously shutdown by the ENGINE FIRE switch (gen field relay open). If generator doesn't connect on line automatically, attempt RESET and ON.

- Verify oil pressure is within starting limits.
- Engine rpms should increase and stabilize at flight idle power. Observe all engine instruments are within limits. Check the respective engine annunciators extinguished.

In Flight Restart — One Engine, Windmilling

- Insure airspeed above 200 KIAS
- Insure N_2 above 8%.
- Altitude below 35,000 feet (Airstart Envelope, *AFM* or checklist)
- Insure throttle — CUTOFF, firewall shutoff — OPEN.
- Place ignition — ON.
- Place Fuel Boost Pump switch — ON.
- Advance throttle — IDLE.
- Monitor engine instruments to insure temperatures, pressures and RPMs do not exceed limits.
- After engine stabilizes, place fuel boost pump switch, NORM and ignition switch, NORM.
- Generator switch, ON.

ENGINE OPERATING CONDITIONS

NORMAL

Taxi

During ground operations, the engines are normally operated with the throttles at idle and the Electronic Engine Controls (EECs) in AUTO. In this configuration, the engines are operating in low speed ground idle (**GND IDLE** annunciator illuminated). Ground idle provides for lower taxi speeds thus reducing brake wear. Power may have to be advanced beyond idle from time to time for uphill taxing and anti-ice/deice systems preflight checks.

Takeoff

Advancing the throttles to the takeoff detents allows the EECs to set takeoff power. It is advisable to consult the N_1 takeoff power charts in the abbreviated checklist by referencing pressure altitude and ram air temperature (RAT gage) to insure the EECs are controlling the engines at the maximum takeoff power setting.

The **GND IDLE** annunciator remains illuminated during the takeoff roll until airborne. The **GND IDLE** light extinguishes as the left squat switch senses the aircraft is airborne.

NOTE

Engines operating normally, EECs AUTO mode, and throttles in the detents, THROTTLE DETENT indicators, adjacent to the N_1 gage, will illuminate as appropriate (Figure 7-13).

Climb

When convenient after takeoff, the throttles may be reduced to the climb detents which alerts the EECs to set Maximum Continuous Multiengine Climb Thrust throughout the climb profile. Select engine sync ON to FAN or TURB as

Figure 7-13 Throttle Detent Indicators

desired. The Multiengine Normal Climb Thrust Setting chart in the abbreviated checklist should be consulted by referencing pressure altitude and corresponding ram air temperature to ensure the EECs are providing proper climb thrust (N_1).

NOTE

The sync switch should be placed OFF for major power changes and single-engine operation.

Cruise

After the aircraft is established at cruise altitude, the throttles may be retarded to the CRUISE detents if maximum cruise power is desired. Consult the two engine cruise charts in Section VII, (FLIGHT PLANNING & PERFORMANCE), of the *Airplane Operating Manual* to reference the N_1 cruise power settings. If reduced power is desired to establish long range cruise or normal cruise power settings, retard the throttles below the cruise detents to establish proper N_1 settings.

Descent and Landing

During descent, reduce throttles as required to establish desired rate-of-descent and airspeed.

Upon landing, there is an eight-second delay before the EECs reduce engine idle rpm to ground idle (GND IDLE annunciator illuminates). The eight-second delay for ground idle is established to allow faster engine spool up time for touch-and-go landings or if an emergency takeoff is desired immediately after touchdown.

Engine Shutdown

Securing the engines is accomplished by retarding the throttles to cutoff, which manually shuts off fuel at the Fuel Control Unit (FCU). The white **EEC MANUAL L/R** advisory lights will illuminate as the throttles are placed in the cutoff position.

EMERGENCY/ABNORMAL

Engine Fire (LH or RH ENGINE FIRE Warning Light On)

On the Ground During Starts. If an engine fire or high ITT occurs during engine start, the throttle should be retarded to CUTOFF and the starter should remain engaged for a minimum of 15 seconds to clear fuel from the engine. Depress the START DISG button to cutoff the starter. If the fire is still present, the affected ENGINE FIRE switchlight should be depressed and then fire both engine fire bottles by pressing both ARM switchlights. After the fire bottles are activated, the ship's electrical power should be secured and the aircraft evacuated.

On the Ground During Shutdown. If an engine fire or high ITT occurs during engine shutdown (throttle cutoff), the respective start button can be pressed to motor the engine in an attempt to clear excess fuel and extinguish the fire. The starter should not be allowed to motor any longer than 15 seconds. Depress the DISG button to terminate starter rotation. If a fire still exists, depressing the corresponding ENGINE FIRE switchlight and firing both armed fire bottles should extinguish the fire.

In Flight. If an ENGINE FIRE light illuminates in flight, reducing the throttle to idle should verify a fire or possibly a bleed-air leak. If the fire light extinguishes after the throttle is reduced, it would normally indicate a bleed-air

leak. In which case, the pilot may elect to perform a precautionary shutdown of the engine or, if required, operate the engine at reduced power.

NOTE

If an engine is shutdown in flight and no fire hazard or engine damage exists, leave the firewall shutoff OPEN and turn fuel boost pump ON to prevent damage to the engine-driven fuel pump.

If the ENGINE FIRE light remains illuminated after the throttle is reduced to idle, it would normally indicate an engine fire. To extinguish the fire, depress the respective ENGINE FIRE switchlight and push an illuminated bottle armed switchlight (either bottle).

Activating an ENGINE FIRE switchlight will close the respective engine fuel and hydraulic shutoff valves (F/W SHUTOFF L or R annunciator illuminated), the generator field relay trips, the respective thrust reverser is disabled and both fire bottles are armed. Depressing either illuminated bottle ARMED switchlight fires an explosive squib on the respective fire bottle and releases fire suppression agent under pressure into the selected engine nacelle (the engine nacelle was selected by pushing the appropriate ENGINE FIRE switchlight). Refer to Chapter 8, FIRE PROTECTION for detailed information regarding the fire detection and extinguishing systems.

Emergency Restart — Two Engines

If both engines should flame out in flight and the airplane has descended below 35,000 feet, placing both ignition switches ON, both fuel boost pump switches ON, and position both throttles to IDLE should cause both engines to restart. If altitude allows, increase airspeed to 200 KIAS. Insure all anti-ice switches are OFF.

If the engines fail to start after approximately ten seconds, attempt starts by momentarily pressing either start button and attempt engine start one at a time.

If the engines still do not start, refer to the **MAXIMUM GLIDE — EMERGENCY LANDING** procedures in the abbreviated checklist, *AFM*, or *Operating Manual*.

Low Oil Pressure (LO Oil PRESS L or R WARNING Light On)

If a **LO OIL PRESS L** or **R** annunciator illuminates “flashing” and the oil pressure indicator displays normal oil pressure in the “green” range, monitor all other engine indicators for any abnormal indications. If all engine indicators reflect normal conditions, the malfunction may have been caused by a faulty low oil pressure switch.

If a **LO OIL PRESS L** or **R** annunciator illuminates “flashing” and the oil pressure indicator displays low oil pressure in the “yellow,” the affected engine throttle should be reduced below 60% N_2, and land as soon as practical.

If a **LO OIL PRESS L** or **R** annunciator illuminates “flashing” and the oil pressure indicator displays low oil pressure below the “red” line (20 psid), the engine should be shutdown. Accomplish abnormal procedure “ENGINE FAILURE/PRECAUTIONARY SHUTDOWN” in the abbreviated checklist, *AFM*, or *Operating Manual*.

Engine Start Malfunction (Engine Does Not Start)

If an engine does not start, placing the throttle in CUT OFF and allowing the starter to remain engaged for approximately 15 seconds, will clear fuel from the engine. Press the START DISG button to terminate power to the starter.

Engine Start Button Light Will Not Extinguish After Engine Start

Automatic Start Sequence Does Not Terminate. If the engine start switchlight does not extinguish after an engine start, and the respective **FUEL BOOST L** or **R** and **GEN OFF L** or **R** annunciators remain illuminated, and the associated ignition light is ON, indicates the generator speed sensor malfunctioned and the start sequence did not terminate. Depress the START DISG switch, located between the start switches. If the start switchlight extinguishes and all other lights associated with the start extinguish, the starter disengage switch was used as a backup to terminate the start sequence. The airplane may be dispatched and operated normally until maintenance can be scheduled at a more convenient time. Further engine starts on the affected engine, may have to be terminated manually until maintenance is performed.

Start Relay Stuck Closed. If after depressing the START DISG switchlight, the affected starter switchlight remains illuminated but other lights associated with the start sequence, i.e., **FUEL BOOST L/R, GEN OFF L/R**, ignition light are extinguished, indicates the start sequence terminated automatically (generator speed sensor functioning properly). The corresponding start relay is stuck.

Prior to shutting down the affected engine, isolate all electrical power sources from the affected starter to prevent the starter from inadvertently motoring the engine during shutdown. If power source isolation is not conducted, as the affected starter/generator voltage decreases during engine shutdown, it will become a starter as electrical power flows through the stuck start relay from the opposite engine generator, battery or GPU. Therefore, place both generators OFF, electrically disconnect the battery by activating the BATTERY DISCONNECT switch on the panel forward of the pilot's circuit breaker panel (battery switch must be ON), and verify a GPU is not connected and shutdown APU if running. After all power sources have been isolated, the affected engine may be shutdown.

NOTE

Open the battery compartment and disconnect the battery prior to turning the BATTERY switch OFF. Placing the BATTERY switch OFF prior to disconnecting the battery will cause the battery disconnect relay to close and provide power to the affected starter and commence motoring the engine.

Oil Filter Bypass (OIL FLTR BP L or R Caution Light On)

An **OIL FLTR BP L** or **R** annunciator that illuminates "flashing," indicates that the respective engine oil is beginning to bypass or has completely bypassed the filter due to possible contamination. Engine oil pressure and temperature should be monitored and consider the possibility of partial or total loss of thrust on the affected engine.

NOTE

A completely blocked oil filter will bypass oil around the Fuel Oil Heat Exchanger (FOHE). Monitor fuel temperature gages and **FUEL FLTR BP L/R** annunciators. If fuel temperature drops considerably, fuel filters could possibly become contaminated with ice.

High Oil Pressure

Above 250 PSI

If oil pressure exceeds the 250 psid red triangle on the oil pressure indicator, the affected engine throttle should be reduced immediately and prepare for landing as soon as practical.

NOTE

Oil pressure at takeoff and maximum thrust settings may exceed 140 psid, not to exceed 250 psid, for up to 120 seconds.

Ground Idle (GND IDLE Advisory Light On)

GND IDLE annunciator illuminated on the ground is normal with the EECs in AUTO.

In Flight. If the **GND IDLE** annunciator illuminates in flight, the ENGINE SYNC should be turned OFF and throttles adjusted to maintain 51.5% N_2 minimum. Engine acceleration in flight from idle may be slower than normal. An illuminated **GND IDLE** light in flight may also indicate a left squat switch malfunction.

Electronic Engine Computer In Manual Mode (EEC MANUAL L or R Advisory Light On)

EEC MANUAL L or **R** annunciator illuminated indicates the respective electronic engine computer is in manual mode. The affected engine throttle should be manipulated to ascertain engine response to throttle movement. If the engine responds, place both EEC switches in MANUAL. In MANUAL mode, the detents will drop out and both throttles can be matched closer to provide equal power. If maximum thrust (N_1) is required, reference the thrust charts in the abbreviated checklist, *AFM* or *Operating Manual*.

If unaffected engine does not respond to throttle in manual, reselect the EEC to AUTO. Throttles will not be matched for equal power.

If affected engine does not respond to throttle in MANUAL mode, select ENGINE ANTI-ICE ON. Required sensing flow should resume, restoring engine response. Verify engine response to throttle movement.

If affected engine responds to throttle with ENGINE ANTI-ICE ON, select both EEC switches to MAN and insure both engine anti-ice switches are ON.

If affected engine still does not respond to throttle (ENGINE ANTI-ICE ON), reduce affected engine throttle to IDLE. Affected engine anti-ice, OFF or AS REQUIRED. Place both EEC switches to AUTO mode and land as soon as practical.

Refer to SINGLE APPROACH AND LANDING checklist. If the affected engine cannot be shutdown with the throttle in cutoff, the respective ENGINE FIRE switch may be pushed.

Engine Vibration (ENG VIB L or R Advisory Light On)

If an **ENG VIB L** or **R** advisory light illuminates, the affected engine vibration monitor has detected a higher than normal level of vibration. The crew should confirm any audible and tactile indications. If a vibration is detected by the crew, monitor the engine instruments for other evidence of malfunction. Consider reducing engine rpm and land as soon as practical.

If vibration increases or other evidence of engine malfunction is present, consider the possibility of shutting down the engine.

CAUTION

IF SIGNIFICANT VIBRATION CONTINUES WITH THE ENGINE RUNNING, ENGINE FAILURE MAY RESULT.

NOTE

Depressing the ENG DATA SCAN button will insure a trace is recorded in the Data Collection Unit to help maintenance troubleshoot the engine malfunction.

LIMITATIONS

ENGINE OPERATING LIMITS

Minimum ambient temperature for ground engine starting is -30°C. See Table 7-1.

ENGINE FAN INSPECTION

To assure accurate fan speed thrust indication, inspect the fan for damage prior to each flight in accordance with the exterior inspection in the NORMAL PROCEDURES SECTION of the *AFM*.

ELECTRONIC ENGINE COMPUTER

Dispatch with either, or both, engines operating in MANUAL mode is prohibited. If it is desired to dispatch in Manual Mode, consult MMEL and Supplement 9 of the AFM.

Engine Sync must be off for takeoff and landing.

Table 7-1 ENGINE OPERATING LIMITS

OPERATING CONDITION		OPERATING LIMITS				
THRUST SETTING	TIME LIMIT (MINUTES)	MAX OBSERVED ITT °C	N_2 %	N_1 %	OIL PRESSURE (NOTE 1) PSI	OIL TEMP °C
TAKEOFF	5 (NOTE 4)	720	101.1	100	45 TO 140	10 TO 121.1
MAXIMUM CONTINUOUS	CONTINUOUS	720	101.1	100	45 TO 140	10 TO 121.1
GROUND IDLE	CONTINUOUS	N/A (NOTE 3)	47 (MIN) (NOTE 2)	_____	45 MINUTES	40 TO 121.1
FLIGHT IDLE			51.5 (MIN)			
STARTING	N/A	720**	---	---	---	-40 MIN
TRANSIENT	20 SECONDS	760**	100*	102*	NOTE 1	121.1 TO 135
TRANSIENT	120 SECONDS	---	---	---	NOTE 1	121.1 TO 135

NOTES:

1. OIL PRESSURE
   - NORMAL OIL PRESSURE IS 45 TO 140 PSID AT N_2 SPEEDS ABOVE 60%. OIL PRESSURE BELOW 45 PSID IS UNDESIRABLE AND SHOULD BE TOLERATED ONLY FOR COMPLETION OF THE FLIGHT, PREFERABLY AT REDUCED POWER SETTING.
   - OIL PRESSURE AT TAKEOFF AND MAXIMUM THRUST SETTING MAY EXCEED 140 PSID (NOT TO EXCEED 250 PSID) FOR UP TO 120 SECONDS.
2. GROUND IDLE IS AVAILABLE IN EEC, AUTO MODE ONLY.
3. FLIGHT IDLE IS AVAILABLE IN EEC, AUTO OR MANUAL MODES.
4. TAKEOFF RATINGS THAT ARE NOMINALLY LIMITED TO 5 MIN DURATION MAY BE USED FOR UP TO 10 MIN FOR ONE ENGINE INOPERATIVE OPERATIONS WITHOUT ADVERSE EFFECTS UPON ENGINE AIRWORTHINESS.

GROUND OPERATION

Continuous engine ground static operation up to and including five minutes at takeoff thrust is limited to ambient temperatures not to exceed 39°C above ISA (Refer to Figure 2-6, *AFM*).

APPROVED OILS

The following oils are approved for use:

- MOBIL JET II
- EXXON TURBO OIL 2380
- AEROSHELL TURBINE OIL 500
- MOBIL JET OIL 254*
- ROYCO TURBINE OIL 500
- AEROSHELL TURBINE OIL 560*
- CASTROL 5000

* Oils denoted with an asterisk are "THIRD GENERATION" lubricants.

CAUTION

WHEN CHANGING FROM AN EXISTING LUBRICANT FORMULATION TO A "THIRD GENERATION" LUBRICANT FORMULATION (AEROSHELL TURBINE OIL 560 OR MOBIL JET 254), THE ENGINE MANUFACTURER STRONGLY RECOMMENDS THAT SUCH A CHANGE SHOULD ONLY BE MADE WHEN AN ENGINE IS NEW OR FRESHLY OVERHAULED. FOR ADDITIONAL INFORMATION ON USE OF THIRD GENERATION OILS, REFER TO ENGINE MANUFACTURER'S PERTINENT OIL SERVICE BULLETINS.

Should it be necessary to replenish oil consumption losses when oil of the same brand (as tank contents) is unavailable, then the following requirements apply:

For contingency purposes, oil replenishment using any other approved oil brand listed is acceptable provided:

1. The total quantity of added oil does not exceed two U.S. quarts in any 400-hour period.

2. If it is required to add more than two U.S. quarts of dissimilar oil brands, drain and flush complete oil system and refill with an approved oil in accordance with *Engine Maintenance Manual* instructions.

Should oils of nonapproved brands or of different viscosities become intermixed, drain and flush complete oil system and refill with an approved oil in accordance with *Engine Maintenance Manual* instructions.

STARTER LIMITATION

Three engine starts per 30 minutes. Three cycles of operation with a 90-second rest period between cycles is permitted.

APPROVED FUELS

The following fuels are approved for use:

- COMMERCIAL KEROSENE JET A
- JET A-1
- JET B
- JP-4
- JP-5
- JP-8 CPW 204 specification

QUESTIONS

1. The primary thrust indicator for PW545A engine is:
   A. Fuel flow
   B. N_1
   C. ITT
   D. N_2

2. If one igniter plug should fail during engine start:
   A. The engine will start normally.
   B. It will result in a "hot" start.
   C. Combustion will not occur.
   D. Engine start will be slower than normal.

3. Ignition during normal engine start is activated by:
   A. Turning the IGNITION switches to ON at 8 to 10% N_2.
   B. Moving the throttle to IDLE at 8 to 10% N_2.
   C. Depressing the start button.
   D. Ignition is not required during normal engine start.

4. Of the following statements concerning the PW545A engine, the correct one is:
   A. Fuel from the engine fuel system is used to cool engine oil through a fuel-oil heat exchanger.
   B. The engine accessory gearbox has its own oil lubricating system (independent of the engine itself).
   C. Fuel is warmed through the fuel-oil heat exchanger.
   D. Both A and C.

5. If the N_1 turbine shaft shifts to the rear:
   A. The engine automatically shuts down.
   B. The vibration detector causes illumination of the master warning lights.
   C. The synchronizer shuts the engine down.
   D. Nothing occurs.

6. The following engine instruments are available in the event of a loss of normal DC electrical power (Emer Power Only):
   A. N_2 rpm
   B. N_1 rpm, N_2 rpm, and ITT
   C. N_1 rpm (tape and LCD display)
   D. N_1 rpm (tape only)

7. The ENGINE SYNC switch:
   A. Should be in FAN for takeoffs and landings.
   B. Should be in TURB at altitude.
   C. Can be placed in FAN or TURB after takeoff and should be left there for the remainder of the flight.
   D. Should be OFF for large power changes.

8. Regarding the Electronic Engine Control (EEC) system:
   A. Place the EEC switch OFF prior to engine start.
   B. Low-speed ground idle is controlled by the EECs.
   C. Engine sync is operational with or without the EEC operational.
   D. If an EEC trips off line in flight, EEC MANUAL L–R annunciator illuminates, the throttle detents will remain operational.

9. During engine starts:
   A. N_1 should register by 25% N_2.
   B. ITT should not exceed 720°C.
   C. Generator assist starts should not be initiated until the operating engine rpm is stabilized at 60% N_2.
   D. Both A and B.

10. Starting engines with a Ground Power Unit (GPU):
    A. Stabilize voltage output 28 VDC.
    B. Amperage output should be adjusted not to exceed 1,200 amps.
    C. Recommend the GPU be plugged into the aircraft before turning on the unit.
    D. After engine starts are complete, place generators online prior to disconnecting the EPU.

CHAPTER 8
FIRE PROTECTION

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FIRE PROTECTION

INTRODUCTION

The Citation Excel is equipped with engine fire-detection and fire-extinguishing systems as standard equipment. The systems include detection circuits which give visual warning in the cockpit and controls to activate one or both engine fire-extinguisher bottles. There is a test function for the engine fire-detection system. Two portable fire extinguishers are stowed inside the cabin.

GENERAL

Each engine fire-protection system is composed of a closed loop sensing system, a control unit located in the tail cone, an ENG FIRE warning switchlight, two fire-extinguisher bottles which are activated from the cockpit, and fire-detection circuit tests. The fire-extinguishing system is a two-shot system; if an engine fire is not extinguished with actuation of the first bottle, the second bottle is available for discharge into the same engine nacelle. The fire bottles are located in the tail cone. Abnormal ambient temperature will also cause the bottles to automatically discharge through relief valves into the tail cone. Selected engine-related systems are automatically shut down upon activation of the fire-protection system by the pilot.

ENGINE FIRE DETECTION AND INDICATORS

SENSING LOOPS AND CONTROL UNITS

Within each engine nacelle is a heat-sensing cable that loops around the lower engine accessory section and the engine combustion section (Figure 8-1). The sensing loops are flexible, stainless steel tubes containing a single wire centered in a semiconductor material. The loops are connected to control units that monitor electrical resistance. As the loop is heated, its electrical resistance decreases until, at a temperature of approximately 500°F, a circuit is completed to the control unit to illuminate the applicable red ENG FIRE switchlight (Figure 8-2). The detection system is powered by main DC supplied through the L and R FIRE DETECT circuit breakers on the pilot's CB panel.

A fault warning (monitoring) system of the fire detection loops is incorporated in the annunciator panel. If the system detects a fault in the sensor loop or the control unit, the amber **FIRE DET SYS L** or **R** annunciator will illuminate.

ENG FIRE AND BOTTLE ARMED SWITCHLIGHTS

The red LH and RH ENG FIRE warning switchlights are located on the glareshield (Figure 8-2). In the event of an engine fire or overheat condition, the applicable warning switchlight illuminates (it **does not** trigger the MASTER WARNING lights). Pressing an illuminated ENG FIRE switchlight arms the fire-

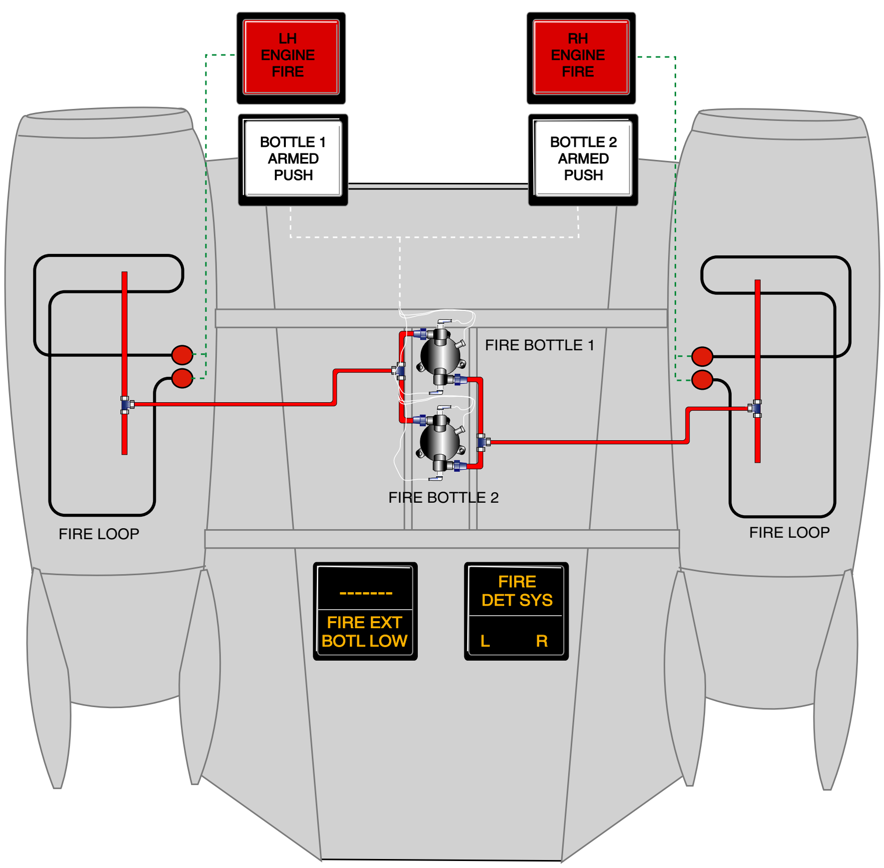

Figure 8-1 Engine Fire System Schematic

Figure 8-2 Fire Warning Switchlight Controls

extinguisher bottles which illuminate both white BOTTLE ARMED switchlights, which are the actuating controls for the fire bottles.

FIRE DETECTION SYSTEM TEST

The rotary test switch, on the center pedestal below the throttle quadrant, is used to test the fire-detection system. Light functionally, and continuity of the sensor loop and detector control units are checked by placing the rotary test switch to FIRE WARN and observing illumination of both red RH/LH ENGINE FIRE lights. If a light doesn't illuminate, the system has a fault.

ENGINE FIRE EXTINGUISHING

EXTINGUISHER BOTTLES

Two spherical extinguishing agent bottles are located in the tail cone area above the baggage compartment area (not visible during preflight inspection) (Figure 8-3). Both bottles use common plumbing to both nacelles, providing the airplane with a two-shot system. The bottles are charged with monobromotrifluoromethane (CBrF3) nitrogen pressurized to 600 psi at 70°F. The extinguishing agent is not corrosive, and its discharge does not necessitate cleaning the engine or nacelle area since it leaves no residue. Release of the extinguishing agent is accomplished by electrical firing of an explosive cartridge on the bottle.

Each bottle incorporates a fusible element that melts at approximately 210°F ambient temperature. If a fire should develop in the tailcone area, contents of the bottles will be released into the tail cone if ambient temperatures increase to 210°F.

NOTE

Due to location of the fire bottles, the bottle pressures cannot be checked in the tailcone during preflight. If either or both fire extinguisher's bottle pressure is low, the amber **FIRE EXT BOTL LOW** annunciator will illuminate to alert the crew.

Figure 8-3 Fire Extinguisher Bottles

OPERATION

An engine fire or overheat condition is indicated by illumination of the applicable ENG FIRE switchlight on the glareshield. After verifying that a fire actually exists, lifting the plastic cover and depressing the illuminated ENG FIRE switchlight causes both white BOTTLE ARMED switchlights to illuminate, arming circuits to the bottles for discharge. In addition, the following occurs:

- The respective generator field relay opens (**GEN OFF L** or **R** annunciator illuminates). The open field relay allows power to close:
- The respective engine fuel and hydraulic firewall shutoff valves (amber **F/W SHUTOFF L** or **R, LO HYD FLOW L** or **R, LO FUEL PRESS L** or **R**, and **FUEL BOOST L** or **R** annunciators illuminate).
- The respective thrust reverser isolation valve is disabled, preventing a possible uncommanded deployment of the thrust reverser (See Chapter 13).

Depressing either illuminated BOTTLE ARMED switchlight fires the explosive cartridge on the selected bottle, releasing its contents into the selected engine nacelle (Figures 8-1 & 8-3). The BOTTLE ARMED switchlight extinguishes and the **FIRE EXT BOTL LOW** annunciator illuminates.

NOTE

> Depressing the ENG FIRE switch light a second time will reopen the firewall shutoff valves, disarm the fire bottles and reset the thrust reversers. *The generator field relay will not reset automatically.*

If the red ENG FIRE switchlight remains on, indicating that the fire still exists, the remaining BOTTLE ARMED switchlight may be depressed, normally after 30 seconds, to release contents of the remaining bottle into the same nacelle. Both detection and extinguishing systems are powered by main DC through the L and R FIRE DETECT CBs located on the pilot's CB panel.

PORTABLE FIRE EXTINGUSHERS

Two hand-held fire extinguishers provide for interior fire protection. Both are 2 1/2 pound Halon 1211 extinguishers, charged with nitrogen at 125 psi. One extinguisher is located under the copilot's seat, the other one is in the aft cabin behind the right rear seat (Figure 8-4).

NORMAL OPERATION

PREFLIGHT CHECK

The engine fire-extinguisher bottles are preflight checked by observing the annunciator panel and ensuring the **FIRE EXT BOTL LOW** annunciator is extinguished. If the annunciator is illuminated, either or both bottles are low or have discharged and maintenance is required prior to dispatch.

Insure the **FIRE DET SYS L/R** annunciators are extinguished. If the annunciator is illuminated, one or both fire detection systems have failed.

ROTARY TEST. Selecting FIRE WARN with the rotary test switch, both red LH and RH ENGINE FIRE switchlights will illuminate indicating a valid fire warning test. If a **FIRE DET SYS L** or **R** annunciator segment is illuminated, the respective fire switchlight will not illuminate during the test.

EMERGENCY/ ABNORMAL OPERATION

ENGINE FIRE DURING TAKEOFF

SPEED BELOW V_1 — TAKEOFF SHOULD BE ABORTED

After the airplane is stopped, push the affected engine fire switchlight and push a BOTTLE 1 or 2 ARMED switchlight. Place ignition NORM and throttle CUTOFF. If fire continues after approximately 30 seconds, fire the remaining fire extinguisher bottle.

SPEED ABOVE V_1 — TAKEOFF SHOULD NORMALLY BE CONTINUED

Figure 8-4 Portable Fire Extinguisher, Copilot's Seat

When clear of obstacles:

ENGINE FIRE (LH OR RH ENGINE FIRE WARNING LIGHT ON)

Reduce the affected engine throttle to idle. If light goes out, normally indicates a possible bleed air leak.

IF LIGHT REMAINS ON

Push the affected illuminated engine fire switchlight and fire either fire extinguisher bottle by depressing a BOTTLE ARMED switchlight.

Secure the engine by accomplishing the following procedures:

- Ignition switch — NORM
- Throttle — CUTOFF
- Electrical load — REDUCE as required (not to exceed 300 amps)
- Affected engine anti-ice — OFF
- WING XFLOW — ON as required (if in icing).

IF FIRE WARNING LIGHT REMAINS ON AFTER 30 SECONDS

Push the remaining BOTTLE ARM switchlight.

Land as soon as possible.

IF FIRE WARNING LIGHT GOES OUT AND SECONDARY INDICATIONS ARE NOT PRESENT

Land as soon as practical.

FIREWALL SHUTOFF VALVE CLOSED (F/W SHUTOFF L OR R CAUTION LIGHT ON)

Indicates the fuel and hydraulic shutoff valves are closed, the generator field relay is tripped, and the respective thrust reverser is disabled.

ENGINE FIRE DETECTION SYSTEM FAILURE (FIRE DET SYS L OR R CAUTION LIGHT ON)

Indicates failure of the affected engine fire-detection system.

ON GROUND

Correct prior to flight.

IN FLIGHT

Check the FIRE DETECT circuit breaker(s) IN on the pilot's CB panel. Monitor the engine instruments for any secondary indications of fire. Land as soon as practical.

NOTE

The fire warning system is inoperative. The firewall shutoff and fire extinguisher bottles are still available if secondary indications of fire are present.

FIRE EXTINGUISHER BOTTLE PRESSURE LOW (FIRE EXT BOTL LOW CAUTION LIGHT ON)

One or both engine fire extinguisher bottles have low pressure or may have discharged.

ON GROUND

Correct prior to flight.

QUESTIONS

1. An ENG FIRE switchlight illuminates:
   A. When it is depressed.
   B. The MASTER WARNING lights also illuminate.
   C. When temperature in the nacelle area reaches 500°F.
   D. Electrical resistance of the sensing loop increases due to increasing nacelle temperature.

2. Depressing an illuminated ENG FIRE switchlight:
   A. Fires bottle No. 1 into the nacelle.
   B. Fires bottle No. 2 into the nacelle.
   C. Fires both bottles into the nacelle.
   D. Illuminates both BOTTLE ARMED switchlights, arming the bottles.

3. After a bottle has been discharged into a nacelle:
   A. No cleaning of the engine and nacelle area is required.
   B. A thorough cleaning of the engine and nacelle area is required.
   C. An inspection of the engine and nacelle area is required to determine if cleaning is necessary.
   D. None of the above.

4. When the fire-extinguishing system is armed for operation (fire switch light depressed):
   A. The LO FUEL PRESS L or R light illuminates.
   B. The LO HYD FLOW L or R light illuminates.
   C. The GEN OFF L or R light illuminates.
   D. All the above.

5. If the contents of an armed bottle has been discharged into a nacelle and the ENG FIRE switch light remains on:
   A. The fire has been extinguished.
   B. The other bottle can be discharged into the same nacelle by depressing the other BOTTLE ARMED switchlight.
   C. The fire still exists, but no further action can be taken.
   D. The same BOTTLE ARMED switchlight can be depressed again, firing a second charge of agent from the same bottle.

6. Depressing the ENG FIRE switchlight a second time:
   A. Opens the fuel shutoff valve.
   B. Opens the hydraulic shutoff valve.
   C. Resets the generator field relay.
   D. A and B above.

CHAPTER 9
PNEUMATIC SYSTEM

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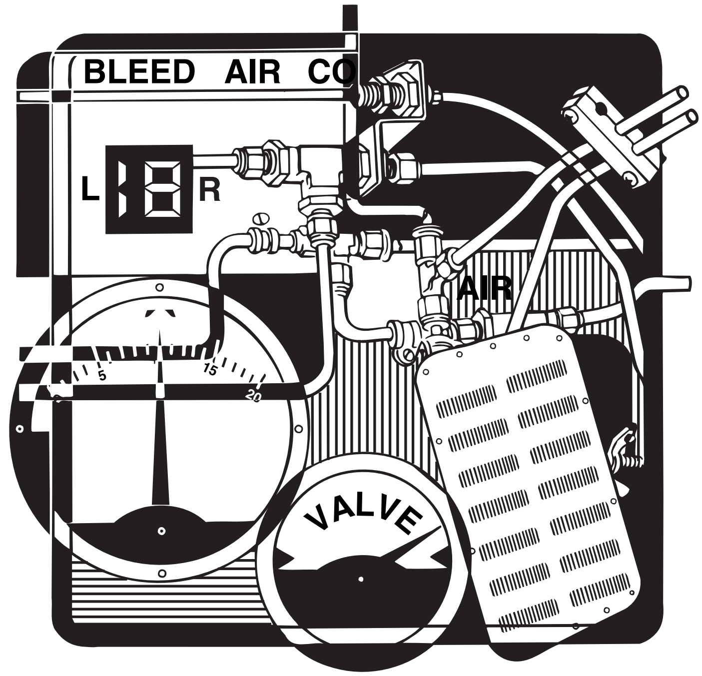
PNEUMATIC SYSTEM



INTRODUCTION

The Citation 560XL pneumatic systems receive HP compressor discharge bleed air (P_3) from the engines and/or optional APU. Bleed air is normally extracted at all times while the engines are operating. The following airplane systems use HP bleed air (P_3):

- Air Conditioning (ACM) and Pressurization (Refer to Chapters 11 and 12).
- Engine inlet anti-ice system (Refer to Chapter 10).
- Wing leading edge anti-ice system (Refer to Chapter 10).
- Horizontal stabilizer deice boot system (Refer to Chapter 10).
- Service air system.

In the event of single-engine operation, bleed air from one engine is sufficient to maintain all required pneumatic systems functions. Safety devices are incorporated to prevent excessive pressures during high power operations. Annunciator CAUTION lights are an integral part of the pneumatic systems.

GENERAL

P_3 bleed air is extracted from the high-pressure (HP) compressors via two discharge ports on each engine. The majority of P_3 air is ducted through the precoolers in each engine pylon

before it is directed into the tail cone. A smaller portion of P_3 air bypasses the precoolers and is ducted directly to each engine inlet anti-ice system. The engine anti-ice system consists of bleed air supplied directly to the engine cowling inlet (lip), and the first and second set of stationary stator vanes behind the fan, one prior to and one directly behind the booster stage (refer to Chapter 10, ICE AND RAIN PROTECTION).

The tailcone bleed air distribution network directs cooled P_3 air to various airplane systems, i.e., air conditioning (ACM), wing leading edge anti-ice, and the service air system.

PRECOOLERS

Precoolers located in each engine pylon (Figure 9-1) receive hot engine bleed air (up to approximately 800°F) and cool it to a level where it is useful for aircraft pneumatic systems. The precoolers are conventional cross flow, air-to-air, heat exchangers. They have two primary air flow paths (hot bleed air flow and cold ram air flow).

Figure 9-1 Pylon Precooler

On the ground, cooling air flow is normally provided by ambient air. If bleed air temperature exiting the precooler exceeds 405°F, a shut-off valve opens and admits engine bypass air for cooling. This normally occurs during high power settings, i.e., during takeoff.

In flight, the engine bypass valve remains closed to conserve air required for thrust. Bleed air is cooled by ram air from a NACA scoop located on the bottom of the pylon (Figure 9-1). Ram air is exhausted through an outlet behind the ram air scoop door (Figure 9-1).

In flight (LH squat switch), the electrically-actuated doors modulate toward open and closed as required to maintain bleed air outlet temperature at 475° ± 25°F. Control of the doors, is accomplished by a sensor located downstream of the precoolers in the bleed air exit ducts. The NACA scoop door is disabled on the ground (closed).

If bleed air temperature exiting the precooler(s) should exceed approximately 560°F, an overheat switch will activate the BLD AIR O'HEAT L or R annunciator flashing.

NOTE

If the wing leading edge anti-ice system is operating, and a **BLD AIR O'HEAT L** or **R** annunciator illuminates, the respective wing anti-ice system shuts down (refer to Chapter 10, ICE AND RAIN PROTECTION).

OZONE CONVERTERS

Two ozone converters mounted in the tail cone are plumbed in such a manner that all engine bleed air is filtered through the converters before entering the Environmental Control Unit (ECU), commonly referred to as the Air Cycle Machine (ACM) (refer to Chapter 11, AIR CONDITIONING). The ozone converters convert ozone to oxygen by catalytic action to enhance air quality at high altitudes.

SERVICE AIR SYSTEM

Service air originates at bleed-air tubes attached to the supply side of the ozone converters in the tail cone. The two bleed-air supplies are routed to check valves and into a cross fitting attached to the 23-psi regulator. From the 23-psi regulator, service air is distributed to the following components (Figure 9-2):

- Horizontal stabilizer deice boots
- Pressurization components
- Cabin entrance door seals
- Throttle detents (EECs AUTO)

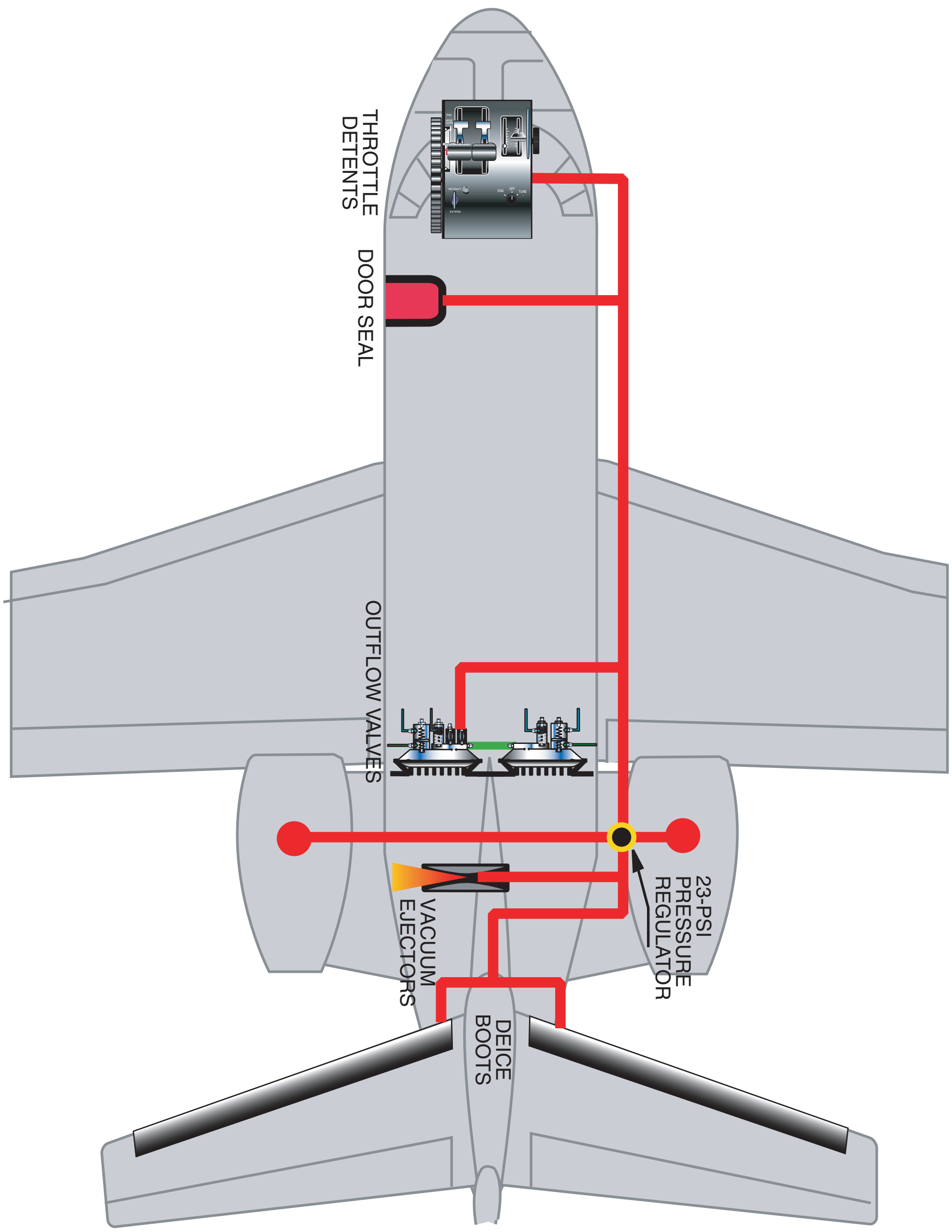


Figure 9-2 Service Air System

HORIZONTAL STABILIZER DEICE BOOTS

Service air (23-psi) is directed to two tail deice control valves. Each valve controls pneumatic air to its respective left or right horizontal stabilizer deice boot. When the system is deenergized, service air passes through the control valve ejectors and creates a vacuum to maintain the boots deflated against the horizontal stabilizer leading edges. Activating the system from the cockpit energizes the control valves to stop service air from flowing through the ejector outlets and directs service air into the deice boot inflation tubes to break off any accumulated ice formation (refer to Chapter 10, ICE AND RAIN PROTECTION).

PRESSURIZATION COMPONENTS

A vacuum ejector supplied with service air is installed on the primary outflow valve. Vacuum generated by the vacuum ejector is applied to the primary outflow valve climb solenoid. During AUTO mode operation of the pressurization system, the climb solenoid valve applies short bursts of vacuum to the outflow valves causing them to slowly open and reduce cabin pressure as required (refer to Chapter 12, PRESSURIZATION).

Service air (23-psi) also passes over an orifice to drop pressure to 6.0 psid over ambient pressure to be directed to the primary outflow valve dive solenoid. During AUTO mode operation of the pressurization system at lower altitudes, the dive solenoid applies short bursts of air pressure at 6.0 psid to the outflow valves causing them to slowly close and increase cabin pressure as required (refer to CHAPTER 12, PRESSURIZATION).

CABIN ENTRANCE DOOR SEALS

Primary Door Seal

The primary door seal consists of a pneumatically inflatable silicone rubber seal installed around the perimeter of the main cabin door. When either or both engines are operating, or an APU, the seal is inflated by 23-psi service air when the door is locked and the locking pins are extended. The lower aft door lock pin actuates a valve that allows bleed air into the door seal. A check valve in the door seal will hold pressure in the seal for a period of time in event of a pneumatic air source loss.

A differential pressure switch, located at the furthest extremity in the inflatable seal from the inflation valve, activates the **DOOR SEAL** annunciator flashing if pressure in the seal drops below approximately 5 psi. The **DOOR SEAL** annunciator will extinguish when the pressure increases to 3 psi above the pressure at which the light illuminates (approximately 8 psi).

NOTE

On the ground with the cabin door closed and locked, prior to first engine start, battery switch ON, the **DOOR SEAL** annunciator will illuminate "steady." If an APU is running, the light will be extinguished.

A vent valve, connected to the primary door seal, is actuated by the door-handle mechanism during the unlocking process to vent air and allow the seal to deflate.

Secondary Door Seal

A bayonet type pressure seal installed around the main cabin door perimeter backs up the inflatable door seal. If the primary door seal deflates, the secondary door seal should seal the door perimeter and prevent cabin air from escaping, thus ensuring cabin pressure integrity.

Acoustic Door Seals

Three acoustic door seals, installed around the main cabin door frame, inflate by service air as the door is closed and locked. A check valve is installed that will prevent the acoustic seals from deflating if pneumatic air pressure is lost. When the door is unlocked, spring-loaded valves

deactivate, shutting off bleed-air pressure and allowing trapped air in the seals to deflate through a vent in the valve body.

NOTE

The acoustic seals do not have annunciator warning lights.

THROTTLE DETENTS

The throttle detents consist of air-operated cylinders and valves that engage or disengage the throttle detent plates as commanded by the Electronic Engine Control (EEC) system. During normal EEC operations in the AUTO mode, the detents will be engaged. EECs operating off line in MANUAL, the detents will be disengaged. The air-operated cylinders are located below the throttle quadrant (refer to Chapter 7, POWERPLANT for more detailed information).

EMERGENCY/ ABNORMAL OPERATION — SERVICE AIR SYSTEM

HORIZONTAL STABILIZER DEICE BOOTS

Refer to Chapter 10, ICE AND RAIN PROTECTION regarding Emergency/Abnormal procedures.

PRESSURIZATION COMPONENTS

Refer to Chapter 12, PRESSURIZATION regarding Emergency/Abnormal procedures.

CABIN ENTRANCE DOOR SEAL — PNEUMATIC

Cabin Door Pressure Seal Failure (Door Seal Caution Light On)

On Ground

Correct prior to flight.

In Flight

Descend to 41,000 feet or lower and land as soon as practical. The secondary pressure seal will maintain cabin pressurization.

PRECOOLER INOPERATIVE

Engine Bleed Air Overheat (BLD AIR O'HEAT L or R CAUTION Light On)

Position the PRESS SOURCE selector switch to the opposite engine and reduce power on the affected engine if practical.

NOTE

If the wing anti-ice system is on and operating, wing anti-ice on the affected engine side will shut down automatically.

THROTTLE DETENTS

If a loss of main DC power occurs or the Electronic Engine Control(s) (EEC) fails, the throttle detents will disengage. Maximum power settings, if required, will have to be maintained manually by referencing power charts in the checklist, *AFM* or *Operating Manual*.

QUESTIONS

1. 23-psi regulated service air provides:
   A. Cabin temperature control.
   B. Wing anti-ice capability.
   C. Pressurization vacuum.
   D. High pressure air to the ACM.

2. Illumination of the DOOR SEAL annunciator is initiated by:
   A. Cabin door seal valve.
   B. <5 psi pressure switch.
   C. Door locking microswitch.
   D. Door handle microswitch.

3. The purpose of service air through vacuum ejector(s) is:
   A. Provide cabin door seal vacuum.
   B. Provide vacuum for the pressurization system and the deice boots.
   C. Provide vacuum for the door acoustic seal.
   D. ACM water separator vacuum.

4. 23 psi service air provides operating pressure for:
   A. Throttle detents and deicer boots.
   B. Standby gyro pressure.
   C. Emergency release of the gear uplocks.
   D. For the wheel brakes accumulator.

CHAPTER 10
ICE AND RAIN PROTECTION

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CHAPTER 10
ICE AND RAIN PROTECTION

INTRODUCTION

The Citation Excel is approved for flight into known icing conditions. The Excel incorporates ice detection, rain removal, anti-icing and deicing systems designed to detect ice and prevent accumulation of ice on critical surfaces. Anti-icing requires that critical surfaces be heated to prevent ice accumulation. This chapter includes information about the systems components, logic and operation.

GENERAL

The Excel utilizes engine bleed air and electrical power to anti-ice and deice critical areas. The surface deice system is designed to remove ice after it accumulates on the leading edges of the horizontal stabilizer, by pneumatically expanding leading edge rubber boots to crack the ice that has accumulated. The air flow moving along the airfoil will then remove the ice.

The surface anti-ice systems will prevent ice accumulation on the protected surfaces. This is accomplished by heating the wing leading edges and the engine inlets with bleed air. The EEC temperature probe (T0), windshield, True Airspeed Temperature (TAS) probe, Angle-of-Attack (AOA), and the pitot/static systems are all heated electrically. The engine fan nose cone and the T_1 (ITT) probe are heated continuously by bleed air during engine operation.

All anti-ice systems should be turned on and operating while in visible moisture and the indicated Ram Air Temperature (RAT) is +10°C or below.

ICE DETECTION

Two red windshield ice-detection barrel lights are mounted on top of the instrument panel glareshield near the windshield and aimed at an area near the inboard edges of each windshield that is not electrically heated (Figure 10-1). When ice begins to form on this area, a small red glow will be reflected on the glass to warn the crew that ice may be accumulating on the aircraft. The windshield ice detection lights are powered anytime the panel light master toggle switch is ON.

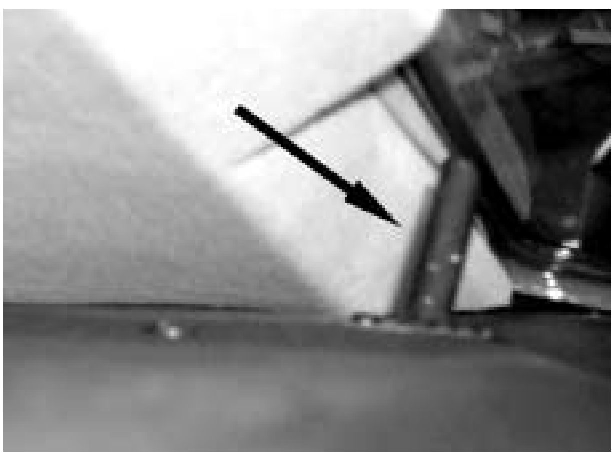


Figure 10-1 Windshield Ice Detection Light

Two wing inspection lights, mounted in each side of the fuselage just forward of each wing root, are used to illuminate the leading edges of the wings during night operations (Figure 10-2). The WING INSP ON–OFF switch is located on the ANTI ICE/DEICE switch panel on the center tilt panel (Figure 10-3).

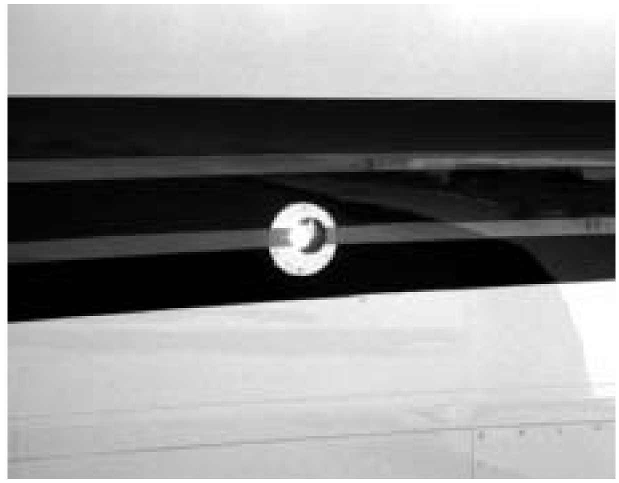


Figure 10-2 Wing Inspection Light

Figure 10-3 Anti-Ice Switch Panel

ANTI-ICE SYSTEMS

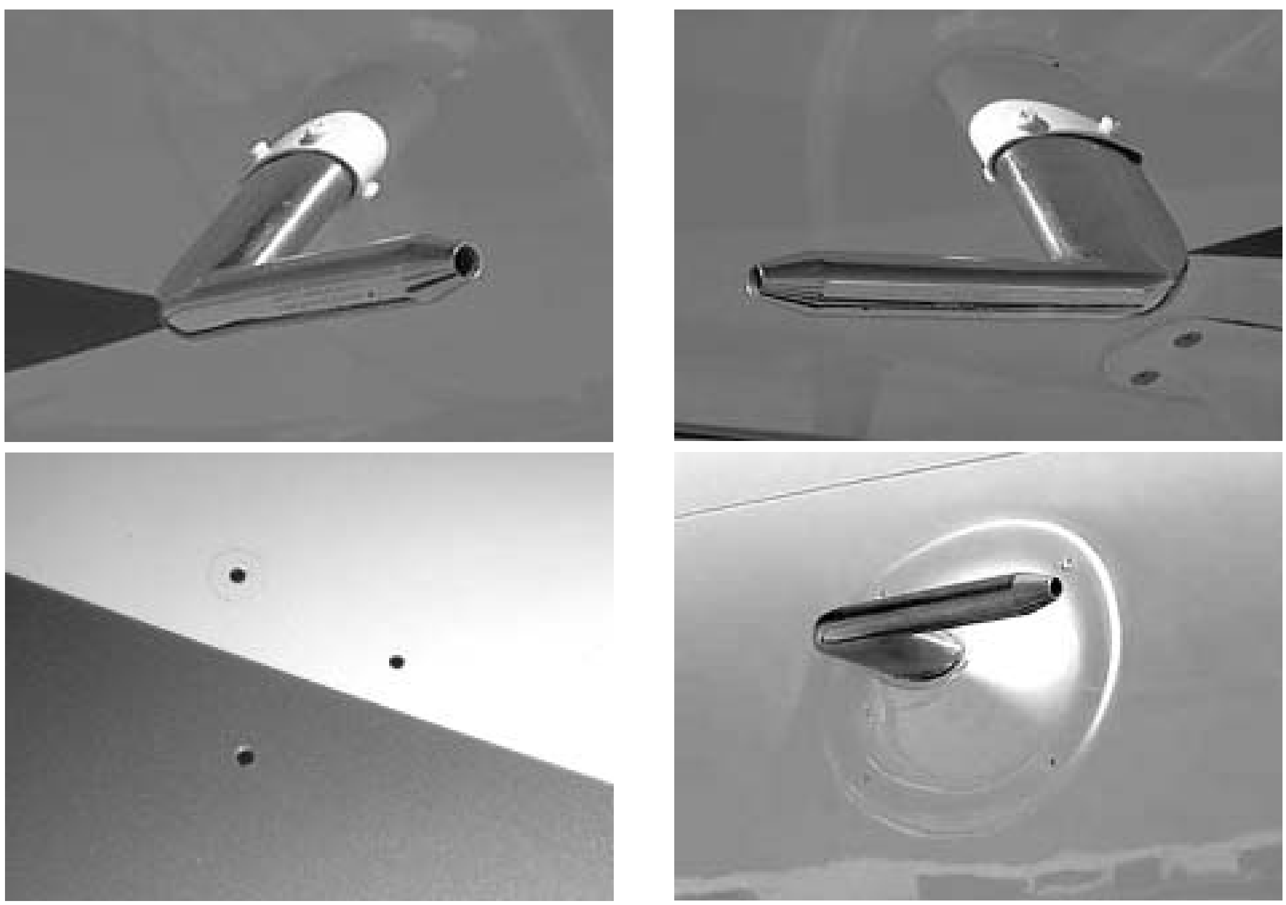
PITOT-STATIC ANTI-ICE SYSTEM

The pitot-static anti-ice system comprises of electrically-heated pitot tubes and static ports. The pitot-static systems include the left and right primary pitot tubes mounted on the nose section of the fuselage, the left and right static port clusters (three static ports each) on each side of

the fuselage below the cockpit, and one standby instrument pitot tube located on the right side of the fuselage below the cockpit (Figure 10-4). The pitot-static anti-ice system is powered by the PITOT & STATIC ON–OFF switch located on the ANTI ICE/DEICE switch panel (Figure 10-3). The warning system consists of electric current sensors which illuminate annunciator panel warning lights in the event a pitot-static anti-ice system heating element fails.

The pitot-static anti-ice system is installed as three independent systems: pilot's system, copilot's system, and the standby system (Meggitt).

The pilot's pitot-static system is the left primary pitot tube, lower forward static port in the left side fuselage cluster, and the upper forward static port in the right side fuselage cluster. Any combination of one or more heating elements that fail in the pilot's system will illuminate the P/S HTR L annunciator "flashing." Conversely, if any combination of one or more heating elements fail in the copilot's system, the P/S HTR R annunciator will illuminate "flashing." The copilot's pitot-static system is the right primary pitot tube, lower forward static port in the right side fuselage cluster, and the upper forward static port in the left side fuselage cluster.



STATIC PORTS

STANDBY

Figure 10-4 Pitot Tubes and Static Port Cluster

The standby pitot-static system includes the right pitot tube located below the copilot's side window and the rear center static ports within each static port cluster on each side of the fuselage. Any combination of one or more heating elements that fail in the standby system will illuminate the STBY P/S HTR annunciator "flashing."

The pilot and copilot's pitot-static anti-ice systems are powered by main DC electric through L and R PITOT STATIC circuit breakers on the pilot's CB panel. The standby anti-ice system is powered from the emergency DC bus through the STBY P/S HTR circuit breaker on the pilot's CB panel. All three pitot-static anti-ice systems, including the angle-of-attack (AOA) vane, are powered ON through the PITOT & STATIC ON–OFF switch.

ANGLE-OF-ATTACK VANE

The angle-of-attack vane (AOA) (Figure 10-5) is heated by placing the PITOT & STATIC switch ON. It is anti-iced electrically by main DC power through the AOA HEATER circuit breaker on the pilot's CB panel. The AOA anti-ice system is monitored by a current sensor and a light on the annunciator panel. If the AOA vane heating element fails, the AOA HTR FAIL annunciator will illuminate "flashing." On the ground, with the PITOT & STATIC switch OFF, the AOA HTR FAIL annunciator will illuminate "steady."

NOTE

When the airplane is on the ground with the PITOT & STATIC ON–OFF switch "OFF," the **P/S HTR L/R, STBY P/S HTR**, and **AOA HTR FAIL** annunciators will illuminate "steady."

CAUTION

LIMIT GROUND OPERATION OF PITOT-STATIC HEAT TO TWO MINUTES TO PRECLUDE DAMAGE TO THE PITOT TUBES AND ANGLE-OF-ATTACK VANE.

TAS TEMPERATURE PROBE

A true airspeed temperature (TAS) probe, commonly referred as the "Rosemount Probe" (Figure 10-6), is mounted on the right side of the nose section to provide temperature data to the two microair data computers (MADC). The probe is heated electrically by main DC through the TAS HEATER on the pilot's CB panel. The probe heater is automatically activated in flight (LH squat switch) with the AVIONICS POWER switch ON. There is no warning light associated with this anti-ice system.

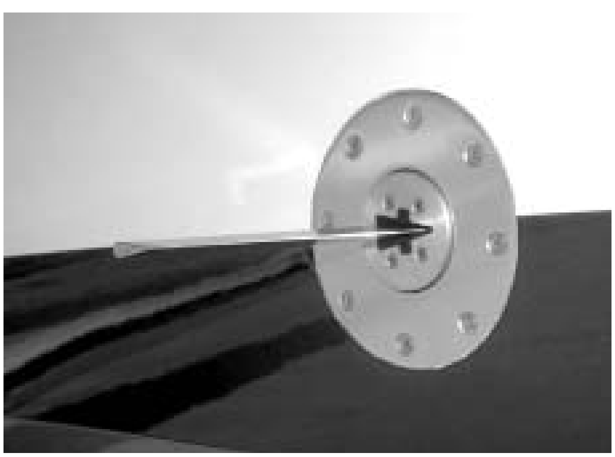


Figure 10-5 AOA Vane

Figure 10-6 TAS Probe

HEATED DRAINS

Electrically-heated drains are provided to prevent ice formation that would impair normal drainage. The left forward refreshment center and cockpit relief tube are equipped with heated drains that operate on main DC power. The heated drains may be located forward, midship, aft or a combination thereof depending on interior configuration. Power is supplied to the heated drains any time power is applied to the airplane (main DC) and the DRAIN HEATERS circuit breaker (located in the aft J-box) is engaged.

WINDSHIELD ANTI-ICE AND RAIN REMOVAL SYSTEMS

General

The windshield anti-ice system consists of electrically-heated glass windshield panels, left and right heated forward glass side windows, electrical control units, associated switches, relays, and annunciator warning lights. The system provides anti-ice and defog capability for the flight compartment windows.

The windshield assembly is laminated all glass construction with bonded fiberglass edge attachments. Heating is accomplished through electrically conductive film applied to the inner surface of the outer glass ply (Figure 10-7). Electrical power is provided by two DC controlled 3.0 KVA AC alternators (one mounted on each engine) supplying 115/200 volt/3-phase, 200 to 400 Hz power. The left and right electric-heated main windshields are divided into three heated zones, each utilizing one phase of AC power. The left and right forward side windows are heated as one section and connected to the main windshields in parallel (Figure 10-8).

Figure 10-7 Electrically-Heated Windshields

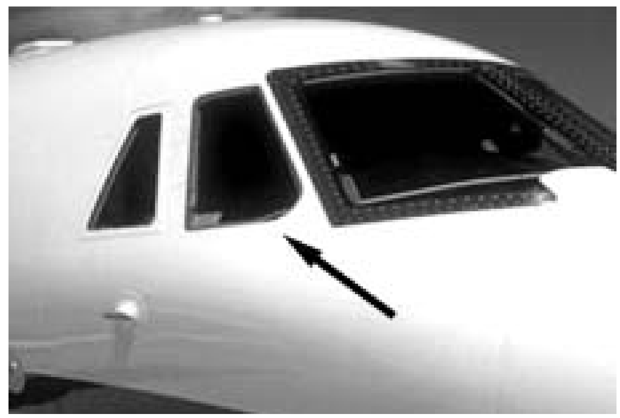


Figure 10-8 Cockpit Forward Side Window

Normal Operations

Two three-position WINDSHIELD L/R OFF–ON–O'RIDE switches located on the ANTI ICE/DEICE switch panel (Figure 10-3) control the system. Placing the switch(es) to the ON position, energizes the respective alternator(s), and initiates ramp heating, regulated by the controllers, to the windshields and forward side windows. Ramp heating heats the glass at a slower rate to desired temperature (reduces thermal shock). If it is desired to heat the windshield more rapidly, the O'RIDE position may be selected.

For normal system operation, the ON position is used.

If icing is encountered with the system OFF, O'RIDE may be used to heat the windshield quicker.

The windshield anti-ice system must be turned ON any time icing is detected. The switches are

normally placed ON prior to engine start and remain ON until shutdown. Windshield heat improves cockpit comfort at high altitude, particularly at night. It is required for windshield defog capability.

AC power for the three windshield zones and the forward side windows are divided as follows (Figure 10-9):

Alternator	Phase	Zones Heated
Left	A	Right W/S inboard zone & Right Fwd Side
	B	Left W/S center zone
	C	Left W/S outboard zone
Right	A	Left W/S inboard zone & Left Fwd Side
	B	Right W/S center zone
	C	Right W/S outboard zone

Two integral temperature sensors are imbedded in each forward windshield center and outboard panel (Figure 10-9). The primary sensors are in the outboard panels and the secondary sensors in the center panels. The sensors are connected to the respective control units to provide constant temperature monitoring to the controllers. The temperatures are regulated at approximately 110°F. Should a fault occur with a sensor, the respective control unit automatically receives temperature input from the remaining sensor. The crew would not notice any change in the system.

Fault Monitoring

W/S O'HEAT L/R and **W/S FAULT L/R** annunciator panel warning lights provide the crew with windshield anti-ice fault monitoring capability.

The **W/S FAULT L** or **R** will illuminate "flashing" if a fault or failure is detected.

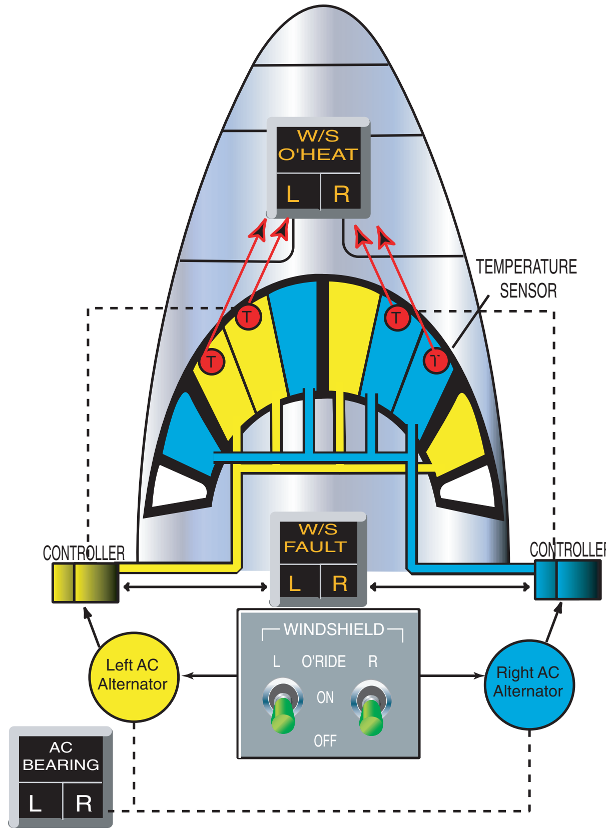


Figure 10-9 Windshield Heat Schematic

Conditions that may trigger a **W/S FAULT L** or **R** annunciator are: Shorted or open circutry/wiring; overheat condition; phase imbalance; controller fault or failure. Illumination of a **W/S FAULT L** or **R** annunciator will cause the respective system to shutdown.

The **W/S O'HEAT L** or **R** annunciator will illuminate "flashing" if an overheat condition (temperature above 140°F) is detected. The system automatically shuts off power to the affected windshield until the overheat condition clears. This condition would normally be caused by a controller failure allowing the windshield to overheat.

If either the primary or secondary sensors detect an overheat condition (above 140°F), the W/S O'HEAT L and/or R annunciator will illuminate

"flashing" and shutoff power to the corresponding windshield anti-ice system. As the respective windshield cools down below 115°F, the respective W/S O'HEAT annunciator will extinguish and power will be restored to the windshield. A corresponding W/S FAULT L or R annunciator will illuminate simultaneously with the W/S O'HEAT L or R annunciator.

NOTE

If the **W/S O'HEAT L** or **R** and the respective **W/S FAULT L** or **R** annunciators cycle ON and OFF, indicating a controller failure, the system may be left ON if needed for icing and defog security, otherwise, the system should be shutdown.

Self Test

The windshield anti-ice system is tested before flight with the rotary test switch located on the center pedestal directly below the throttle quadrant. The system is tested with the WINDSHIELD switch ON and rotating the TEST switch to W/S TEMP.

Proper test indications with the engines shutdown:

- The **W/S FAULT L/R** annunciator should illuminate continuously. The **W/S FAULT L/R** light detects an alternator fault, because the alternator(s) are not operating with the engine(s) shutdown. The **W/S O'HEAT L/R** annunciators will illuminate for 3 to 4 seconds and extinguish.

Proper test indications with the engines running:

- The **W/S FAULT L/R** and **W/S O'HEAT L/R** annunciators should illuminate for 3 to 4 seconds and extinguish. Indicates the controller is functional, AC alternator three phase circuitry is balanced, and the primary and secondary sensors are intact. If a **FAULT** light remains illuminated, may indicate alternator failure, controller failure, or a primary or secondary sensor failure.

Rain Removal System

The glass windshield is treated with a permanently coated and sealed rain repellent agent that is effective during flight in rain.

CAUTION

DO NOT APPLY UNAUTHORIZED RAIN REPELLENT COATING OR COMPOUNDS TO ELECTRIC HEATED GLASS WINDSHIELD OR ASSOCIATED HEATED GLASS SIDE WINDOWS. SURFACE SEAL (TM) IS THE ONLY AUTHORIZED RAIN REPELLENT COATING. APPLY ONLY WITH WINDSHIELD MANUFACTURER AUTHORIZATION AND INSTRUCTIONS.

On the ground, the rain removal system consists of a nose compartment mounted two-speed blower fan and shroud assembly that directs airflow across the surface of the windshield. The fan is effective for rain removal on the ground only. The fan is controlled by the WINDSHIELD AIR ON–OFF switch located on the ANTI ICE/DEICE switch panel (Figure 10-3). Placing the control switch ON, the blower motor runs in the high speed rain removal mode.

With the switch OFF, the fan is not normally operating. However, an overtemperature switch is installed in the nose compartment, and if nose compartment temperature exceeds approximately 90°F, the switch closes and causes the blower to operate at low speed. This is the avionics cooling mode and exhausts hot air out of the nose compartment.

ENGINE ANTI-ICE OPERATION

General

Engine anti-icing includes the bleed air heated T_1 (ITT) probe and the fan assembly nose cone (Figure 10-10). They are heated continuously when the engine(s) are operating.

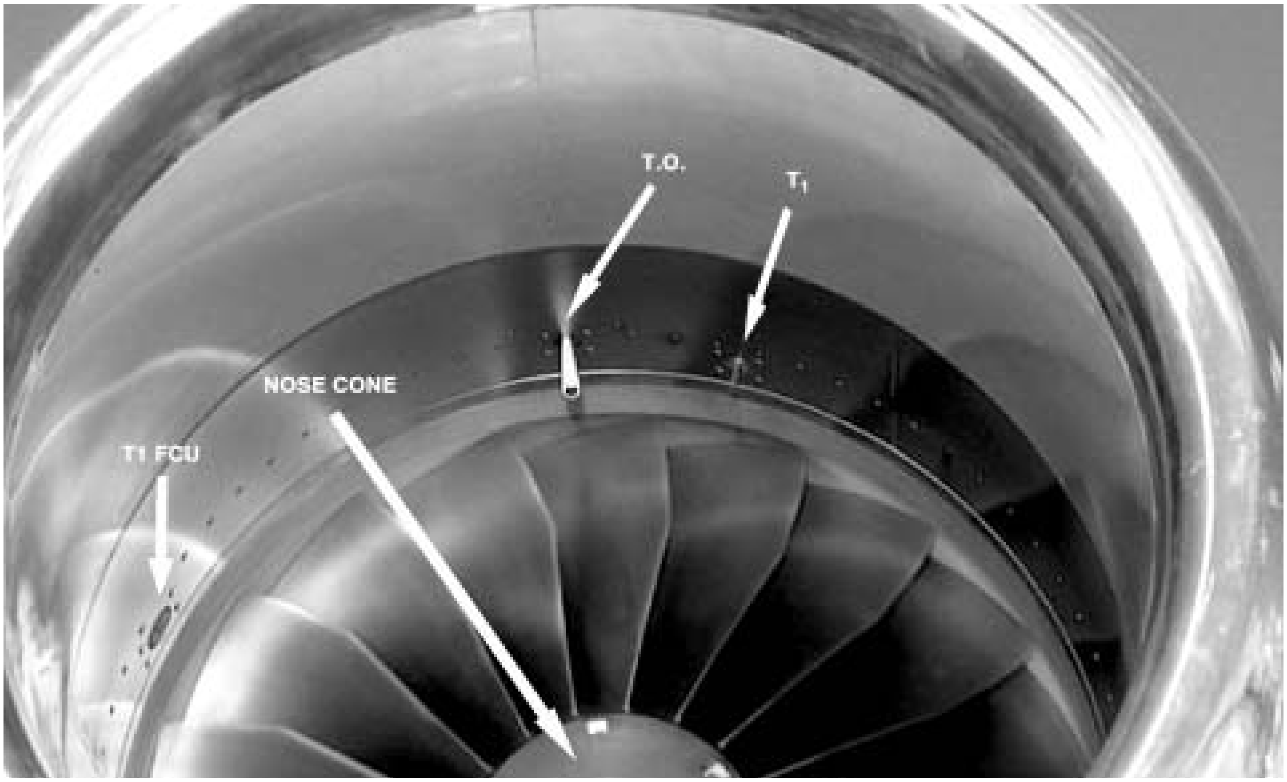


Figure 10-10 Engine Inlet

Selecting ENGINE ANTI-ICE ON allows the following (Figure 10-11):

1. Engine P_3 bleed air to heat the nacelle inlet (lip) including the T1 FCU.
2. Engine P_3 bleed air to heat the two sets of engine stators associated with the fan assembly.
3. Activates ignition automatically.
4. Provides electrical heat for the Electronic Engine Control (EEC) T.O. temperature probe.

Normal Operations

Bleed air for engine inlet anti-icing is supplied directly from the engine being anti-iced. This air is not precooled. Two solenoid-controlled shutoff valves mounted on each engine are controlled by two three-position ENGINE ON–OFF–WING/ENGINE ON switches located on the ANTI ICE/DEICE switch panel (Figure 10-3). Placing the switch(es) to either the ENGINE ON or WING/ENGINE ON position will deenergize the valves and allow P_3 bleed air to flow to the nacelle lip and to the fan case stator vanes.

NOTE

There are no provisions for cross-flow of bleed air between engines if one engine bleed air anti-ice system malfunctions.

Activating the engine anti-ice system by placing the ENGINE ANTI-ICE switches to either ENGINE ON or WING/ENGINE ON position, will automatically activate ignition (ignition lights illuminate) and apply electrical heat to the EEC T.O. temperature probe(s).

If the throttles are in the EEC detents, power will automatically compensate for anti-ice on.

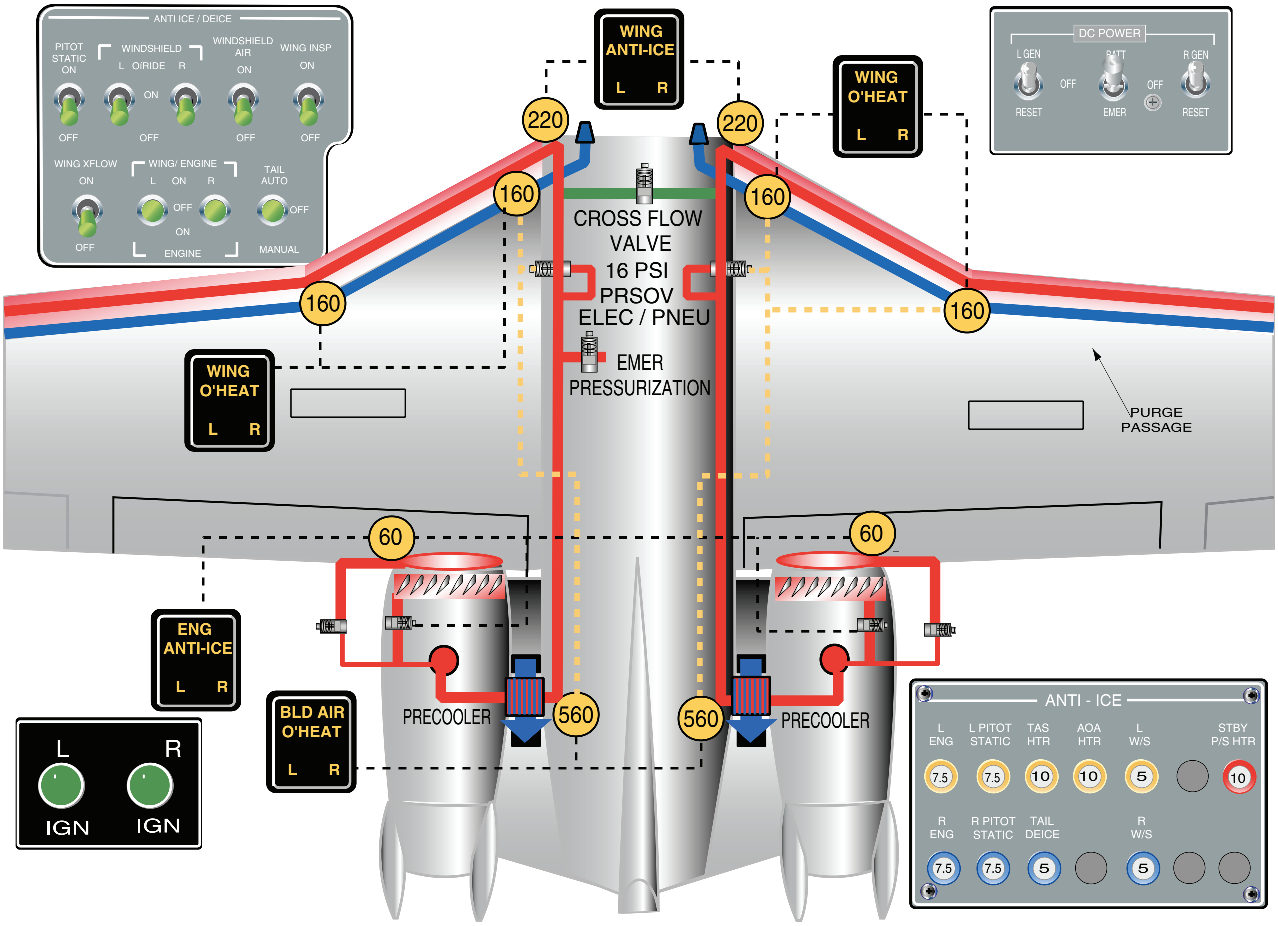


Figure 10-11 Wing/Engine Anti-Ice Schematic

NOTE

The T.0. temperature probe mounted at 12 o'clock in the RH engine nacelle inlet provides temperature input to the Ram Air Temperature indicator on the pilot's instrument panel (Figure 10-12). If the T.0. probe fails, the No. 2 Microair Data Computer (MADC) will provide temperature to the RAT indicator.

Figure 10-12 Ram Air Temperature Indicator

Engine Inlet Anti-ice

The engine nacelle inlet assembly consists of a circular plenum with a circular piccolo tube that fits just behind the forward surface of the inlet. Bleed air enters the piccolo tube at the top and impinges on the forward surface of the plenum transferring heat to the inlet case, then travels aft in the plenum and exhausts out the rear of the engine inlets.

Stator Vane Anti-ice

The stator vane shutoff valve opens as the ENGINE ANTI-ICE switch(es) are positioned to ENGINE ON or WING/ ENGINE ON supplying P_3 bleed air to the two rows of fan case stator vanes. The stators vanes are hollow to allow P_3 bleed air to flow through. The first set of stators aft of the fan, directs air from the fan to the booster stage, and the second set of stators aft of the booster, directs air to the axial flow compressors.

Annunciator Warnings

Engine inlet under temperature switches monitor temperature of the air in the nacelle inlet. When air temperature in the inlet plenum is less than 60°F the **ENG ANTI-ICE L** and/or **R** annunciator will illuminate. Initially turning the system ON, the **ENG ANTI-ICE** annunciators will illuminate "steady" until temperature in the inlet reaches operating temperature (60°F) and then extinguishes.

In flight, if operating temperature is not achieved within 4 minutes and 45 seconds, the annunciator(s) will commence "flashing." Nuisance trips (less than five seconds between resumption of normal temperature and the detection of a new undertemperature condition) are inhibited by annunciator circuit logic.

In flight, if the stator valve does not open when the engine anti-ice system is activated, the **ENG ANTI-ICE L** and/or **R** annunciator will illuminate "flashing" after 4 minutes and 45 seconds.

If the stator valve does not close after selecting the engine anti-ice system OFF, the **ENG ANTI-ICE L** and/or **R** annunciator will illuminate "flashing."

On the ground, selecting the engine anti-ice systems ON will cause the **ENG ANTI-ICE L** and/or **R** annunciators to illuminate "steady" only. The lights will extinguish when operating temperature is reached in the nacelle lip and the stator vane valve is open.

NOTE

The valves for the nacelle inlet and the stators will fail open if a loss of main DC electrical power occurs.

WING ANTI-ICE OPERATION

General

The wing anti-ice system uses high-pressure engine bleed air (P_3) to warm the leading edges to prevent ice accumulation. The system incorporates the precoolers installed in the engine pylons, Pressure Regulating Shutoff Valves (PRSOV), one for each wing, a cross flow plenum with a cross flow valve, two wing leading edge assemblies, various temperature switches, cockpit control switches, and annunciator warning lights (Figure 10-11). Each engine supplies bleed air to its respective wing. However, a cross flow system is incorporated to allow one engine to supply bleed air to both wings during single-engine operation if environmental conditions warrant.

Normal Operations

Bleed air from the engines flowing through the precoolers into the tail cone bleed air ducts, is restricted from reaching the wing leading edges by the wing anti-ice Pressure Regulating Shutoff Valves (PRSOV). The valves are energized closed with the WING/ENGINE switch OFF (Figure 10-3). The wing anti-ice system is placed in operation by positioning the WING/ENGINE switches ON. Bleed air flows from the tailcone forward along the fuselage above each wing through the PRSOVs to the wing anti-ice manifold assemblies at the wing root leading edge. Bleed air then flows into the wing leading edge assemblies, which are piccolo tubes that distribute hot bleed air along the leading edges (Figure 10-13).

NOTE

The wing anti-ice Pressure Regulating Shutoff Valves (PRSOV) regulate bleed air pressure at approximately 16 psi.

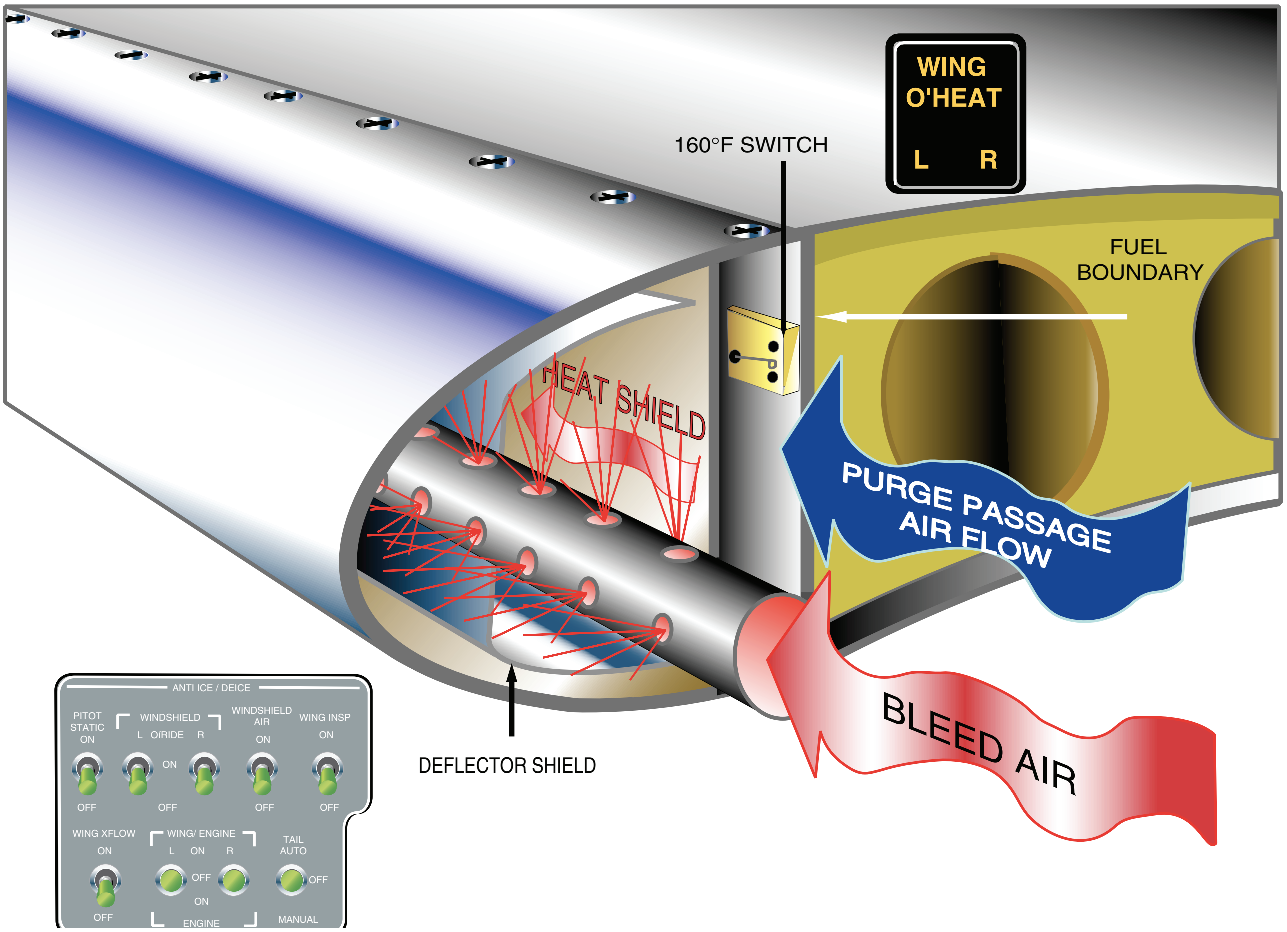


Figure 10-13 Wing Leading Edge Cross Section

Undertemperature switches located in the wing root area monitor the temperature of the bleed air entering the wing leading edge anti-ice panel assemblies. If the bleed air temperature is less than approximately 220°F, the **WING ANTI-ICE L** and/or **R** annunciator will illuminate. The annunciator will illuminate "steady" when the system is initially turned on until air temperature in the wing root increases to operating temperature (220°F or higher) and then extinguishes. If normal operating temperatures are not reached within 4 minutes and 45 seconds, the respective annunciator will commence "flashing." Nuisance trips (less than five seconds between resumption of normal temperature and the detection of a new undertemperature condition) are inhibited by annunciator circuit logic.

NOTE

The precoolers are designed to maintain proper bleed air temperature for the wing anti-ice system.

An undertemperature condition may be caused by the following conditions:

- Insufficient bleed air flow
- Leakage in bleed air lines
- Malfunctioning controlling components
- Improper air circulation at the leading edge anti-ice panels
- Electrical malfunctions
- Low engine rpm combined with low ambient temperature conditions.

Wing Supply System

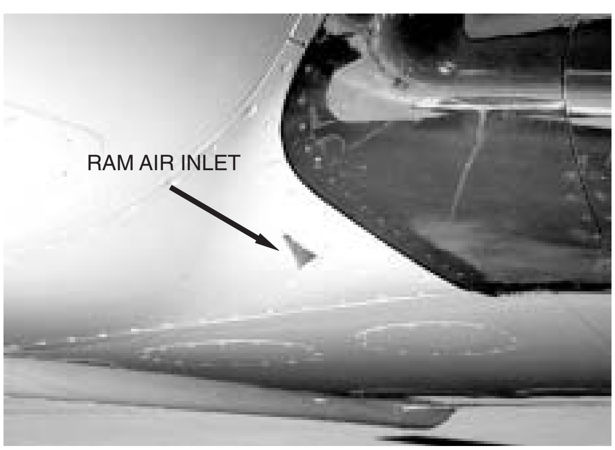
The wing leading edge assemblies consist of inboard and outboard stainless steel leading edge assemblies with inner diffusers, heatshields and piccolo tubing (Figure 10-13). The piccolo tubes have holes drilled at various spacing and angles to provide proper bleed air heat distribution to the wing leading edge. The anti-ice panel assembly is divided into two distinct chambers.

In the forward chamber, bleed air is supplied to the inboard and outboard piccolo tubing which runs the entire length of the wing leading edge. Bleed air exits the piccolo tube and impinges directly on the leading edge. An inner liner (deflector shield) directs bleed-air flow near the leading edge to extract the maximum amount of heat possible. Spent bleed air is discharged from the wing leading edge cavities through overboard vents in the lower surface of the outboard wing (Figure 10-14). The second chamber is located between the fuel bays and the first chamber heat shield and is vented by a ram air vent inlet located on the bottom side of the wing root area (Figure 10-14). This chamber prevents hot bleed air or fuel vapors from accumulating. Ram air is kept separate from the bleed air chamber. Ram air flows outboard through the vent chamber and exhausts through an overboard vent on lower wing surface near the wing tip (Figure 10-14).

Overheat Indications

Two wing overtemperature switches mounted on the forward wing spar in the ram air vent chamber, one located near the wing root and one located near the junction where the inboard and the outboard leading edge panels join, are set to close when the temperature in the chamber exceeds 160°F. At this temperature, when the switch(es) close, the **WING O'HEAT L** and/or **R** annunciator will illuminate "flashing" and the respective PRSOV valve will close shutting off bleed air to the overheated wing panel. The annunciator will extinguish as the temperature drops below 160°F and allow the respective PRSOV valve to reopen. Overtemperature indications may be caused by bleed-air leaks at monitored locations. Bleed-air leaks may cause the **WING O'HEAT L** and/or **R** annunciator to periodically cycle ON and OFF and the associated PRSOV valve(s) to cycle open and closed.

If the precooled bleed-air temperature exiting the precooler(s) exceeds approximately 560°F, the **BLD AIR O'HEAT L** and/or **R** annunciator will illuminate "flashing" and the respective PRSOV valve will close shutting off bleed air and protect the wing. If bleed air from the precooler returns to normal, the annunciator will extinguish and the PRSOV valve(s) will reopen.



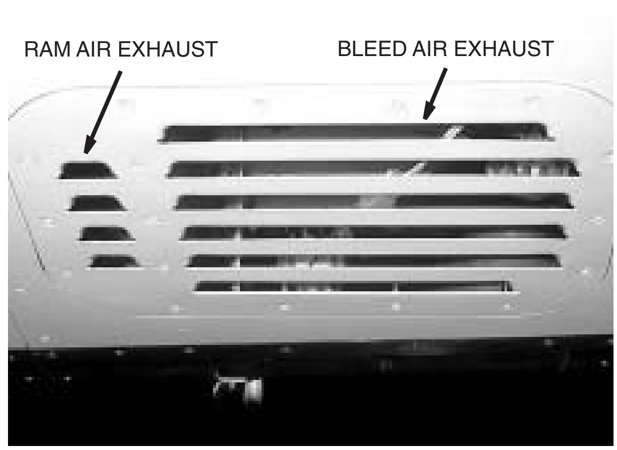


Figure 10-14 Wing Ram Air Inlet and Exhaust Vent

Cross Flow

Downstream from the pressure-regulating shutoff valves, the two wing bleed-air supply lines are connected by a crossflow line. The crossflow line and affixed shutoff valve allow one engine to supply bleed air to both wings (Figure 10-11).

The crossflow shutoff solenoid valve is controlled by a two-position WING XFLOW ON–OFF switch located on the cockpit ANTI ICE/DEICE switch panel (Figure 10-3). Placing the switch ON, in conjunction with the WING/ENG ANTI-ICE L or R switch ON, applies electrical power to open the valve. Placing the switch OFF, deenergizes the valve closed. During single-engine operations or a loss of bleed air supply to a wing (**BLD AIR O'HEAT L** or **R** light will not extinguish), activating WING XFLOW ON, allows the functioning wing side to supply anti-icing capabilities to both wing leading edges.

NOTE

Selecting WING XFLOW ON will not affect annunciator lights for engine anti-ice and/or wing anti-ice on the receiving side.

NOTE

When operating at or above Maximum Continuous Thrust, and RAT is approximately 0°C or warmer, selection of WING/ENGINE anti-ice ON may cause a momentary amber BLD AIR O'HEAT annunciation and illumination of MASTER CAUTION. This situation is not hazardous and will correct itself in a few seconds.

CAUTION

OPERATING THE SYSTEM ON THE GROUND ON AN EXTREMELY HOT DAY, WITH ENGINES AT A SETTING OF OVER 70% N_2 OR GREATER, MAY CAUSE AN OVERTEMPERATURE INDICATION EVEN THOUGH THERE IS NO SYSTEM FAILURE.

OPERATE THE SYSTEM ONLY LONG ENOUGH TO SEE THE ANNUNCIATOR LIGHTS EXTINGUISH, THEN SHUT THE SYSTEM DOWN. CONTINUED OPERATION OF THE SYSTEM MAY CAUSE DAMAGE TO THE HEATED PANELS.

RAM AIRFLOW IS NOT AVAILABLE TO PRECOOL THE ENGINE BLEED AIR DURING GROUND OPERATION. ENGINE OPERATION ABOVE APPROXIMATELY 70% N_2 CAN TRIP THE BLD AIR O'HEAT L AND/OR R ANNUNCIATOR ON. THE ENGINES SHOULD NOT BE RUN ABOVE 70% N_2 FOR GREATER THAN ONE MINUTE UNLESS THE BLEED SYSTEMS (ENVIRONMENTAL AND ANTI-ICE SYSTEMS) ARE SELECTED OFF.

DEICE SYSTEM

HORIZONTAL STABILIZER DEICE

General

The tail deice system for the horizontal stabilizer is a pneumatic boot system. Rubber boots are bonded to the horizontal stabilizer leading edges (Figure 10-15). Engine bleed-air pressure is used to inflate the boots and break the accumulated ice buildup, and airflow across the stabilizer removes the cracked ice from the boots.

Major components of the tail deice system are: a control switch in the cockpit, a timer/logic PC board, two control valves, two pressure switches, two rubber deice boots, and deice annunciators.

Figure 10-15 Horizontal Stabilizer Deice Boots

Service air (23 psi) pressure is used to inflate the boots and to provide a source of vacuum to draw the boots down and hold them tight against the horizontal stabilizer leading edges.

Operation

The TAIL AUTO–OFF–MANUAL switch located on the ANTI ICE/DEICE switch panel controls the deice system (Figure 10-3). The system is powered by main DC through the TAIL DEICE circuit breaker on the pilot's CB panel. The system is completely deenergized in the OFF position. In this position, 23-psi service air is allowed to flow through vacuum ejectors built into the control valves (Figure 10-16).

Vacuum created through the vacuum ejectors hold the boots firmly against the leading edges of the horizontal stabilizer. Exhaust from the vacuum ejectors is directed into the tailcone area. In AUTO, the timer/logic PC board controls an automatic inflation/deflation cycle of the boots. MANUAL position is used as an override backup mode if the AUTO mode malfunctions.

NOTE

Engine rpms greater than 60% N_2 are normally required to develop enough bleed-air pressure (service air) to adequately inflate the boots.

Auto Mode

The pneumatic boots are normally operated by placing the TAIL AUTO–OFF–MANUAL switch to AUTO. AUTO selection activates the timer/logic PC board to start an 18-second inflation/deflation cycle (Figure 10-16).

During the first six second interval (1 to 6 seconds), the left control valve closes preventing service air from flowing through the vacuum ejector and directs 23-psi air to the left horizontal stabilizer boot allowing it to inflate. During the next six second interval (7 to 12 seconds), the left control valve opens allowing vacuum to deflate the left boot. The last six seconds (13 to 18 seconds), the right control valve closes preventing service air from flowing through the vacuum ejector and directs 23-psi air to the right horizontal stabilizer boot allowing it to inflate.

After the last six second interval has expired, the timer rests for approximately three minutes and the 18-second cycle repeats itself. With the control switch in the AUTO position, the system continually repeats the 18-second inflation/deflation cycle every three minutes.

The system is designed to operate in AUTO and remain in AUTO while operating in an icing environment or anytime airframe icing is suspected. The cycling action every three minutes will not allow enough ice to build up to

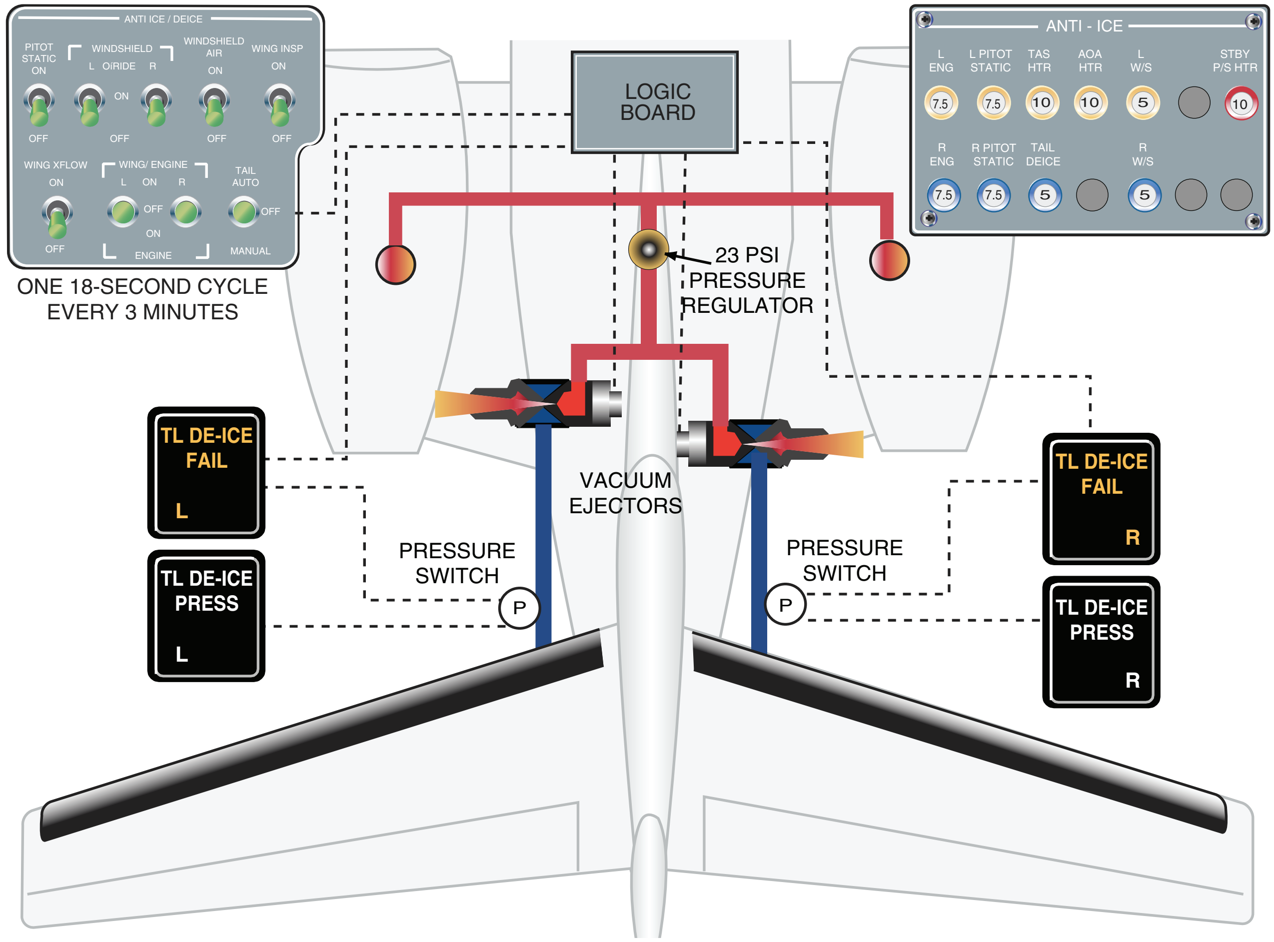


Figure 10-16 Horizontal Stabilizer System Schematic

prevent the boots from breaking the ice nor allow the ice to "bridge."

Manual Mode

Placing the control switch to MANUAL bypasses the timer logic and applies DC power to both control valves. This action shuts off vacuum and allows 23-psi service air to both boots simultaneously. The boots will remain inflated as long as the control switch is held in the MANUAL position. Releasing the switch allows the switch to spring back to the OFF position. Manual operation requires main DC electrical power.

CAUTION

OPERATING THE SYSTEM IN MANUAL WHILE IN ICING CONDITIONS, REQUIRES ATTENTION. HOLD THE SWITCH IN MANUAL FOR APPROXIMATELY 6 TO 8 SECONDS AND RELEASE. REPEAT AT 3 TO 5 MINUTE INTERVALS.

Fault Warnings and Monitoring

The tail deice system is monitored by an "amber" **TL DEICE FAIL L/R** annunciator and a "white" **TL DEICE PRESS L/R** advisory light in the annunciator panel.

Two pressure switches are installed in the pneumatic deice lines. They are installed inside the vertical stabilizer, one each on the left and right side below the horizontal stabilizer. They provide indication of proper operation of the deice boots. The pressure switches operate at approximately 12-16 psi. As the boots inflate with sufficient pressure, the respective switch(es) actuate the “white” TL DEICE PRESS L and/or R advisory light “steady.”

During AUTO operation, the **TL DEICE PRESS L** light will illuminate for a few seconds during the first cycle, and then extinguish during the second (rest) cycle. The **TL DEICE PRESS R** light will illuminate for a few seconds during the last cycle (third) and then extinguish.

As the cycle repeats after each three minute rest period, the advisory lights will illuminate again as previously described until the control switch is placed OFF.

During MANUAL operation, both “white” advisory **TL DEICE PRESS L** and **R** lights will illuminate simultaneously as both boots inflate. They remain inflated as long as the switch is held in the MANUAL position.

The “amber” **TL DEICE FAIL L** and/or **R** annunciator will illuminate “flashing” if the control switch is in AUTO and boot inflation pressure did not reach operating pressure or the boots did not cycle properly (timer or control valve failure). The **TL DEICE FAIL L** and/or **R** annunciator will illuminate “flashing” with the control switch OFF if the timer and/or control valve(s) are still energized.

NOTE

The **TL DEICE FAIL L/R** annunciator is disabled with the deice control switch in MANUAL. The white **TL DEICE PRESS L/R** advisory lights are still active in MANUAL.

NORMAL OPERATION

PREFLIGHT

Exterior Inspection

Conducting the EXTERIOR INSPECTION, all pitot tubes, static ports and Angle-of-Attack vane are checked CLEAR and WARM during the Hot Items/Lights check. The TAS probe is checked to ensure it is clear. The TAS probe heater activates when airborne with the avionics switch ON.

The engine T_1 and T.0. probes are inspected for damage. The wing inspection lights are checked for cracked lens and security. The leading edge of the wings are checked for condition and all vents clear (anti-ice bleed air cooling inlet near the wing root and the bleed air/ram air exhaust vents on the lower surface of the wing toward the wing tip).

The horizontal stabilizer deice boots are checked for any evidence of tears, cracks, delaminating, etc.

Cockpit Preparation

The windshield anti-ice switches are normally placed ON during the Cockpit Preparation checks and left on continuously.

The windshield anti-ice system is checked with the system ON during the Warning Systems Rotary Test. Selecting W/S TEMP position the **W/S O’HEAT L/R** annunciator will illuminate for a few seconds and extinguish. The **W/S FAULT L/R** annunciator will illuminate and remain on while the test switch is in the W/S TEMP position. The alternators are inoperative with the engines shutdown causing the **W/S FAULT L/R** annunciators to remain illuminated during the test function.

NOTE

With the WINDSHIELD ANTI-ICE switches ON and the engines running, a defective alternator will be indicated by the respective W/S FAULT L or R annunciator illuminating “flashing.”

NOTE

The **W/S FAULT** annunciator may not test after cold soak at extremely cold temperatures. If this occurs, repeat the test after the cockpit has warmed up. The test must be completed prior to flight.

If the windshield is heat soaked above 56°C (134°F), the test will result in a W/S FAULT annunciator.

Starting Engines

After the engines are started:

NOTE

When operating in visible moisture and ambient air temperature is +10°C or below, turn pitot and static heat ON and engine L and R anti-ice systems ON.

Before Taxi

Icing conditions exist when the indicated RAT on the ground and for takeoff is +10°C or below and visible moisture is present, and operating on ramps, taxiways or runways where snow, ice, standing water, or slush may be ingested by the engines or freeze on engine nacelles or engine sensor probes.

Conducting the BEFORE TAXI checklist, the anti-ice/deice systems are checked and set as required.

Place engine/wing anti-ice switches to WING/ENGINE ON position and observe a decrease in engine rpm and an increase in ITT (may be a small momentary rpm decrease (N_1) and a slight ITT increase if ENGINE ON only is selected).

Observe the **ENG ANTI-ICE L/R** and **WING ANTI-ICE L/R** annunciators illuminated.

NOTE

If ambient temperature is approximately +15°C or warmer, the **ENG ANTI-ICE L/R** annunciators may not illuminate when anti-ice is selected ON. To insure bleed air is flowing to the engine inlet, observe a momentary small decrease in N_1 with ENGINE ON only.

If sufficient bleed air flow is not available to maintain proper wing temperature, the **WING ANTI-ICE L/R** annunciators will remain illuminated. The lights may be extinguished by increasing engine rpm. It may require up to two minutes to extinguish the **WING ANTI-ICE L/R** annunciators with N_2 set at approximately 70%.

NOTE

During sustained ground operations in freezing precipitation, the engines should be operated for 15 seconds out of every four minutes at 65% N_2 or above to preclude ice forming on engine probes or internal components.

CAUTION

DURING SUSTAINED GROUND OPERATIONS IN FREEZING PRECIPITATION, IF THE ENGINES ARE OPERATED AT IDLE, ICE MAY FORM ON ENGINE PROBES AND INTERNAL COMPONENTS. THIS MAY CAUSE ENGINE VIBRATION AND ERRONEOUS RAT INDICATIONS. BY INCREASING ENGINE SPEED TO 60% N_2 OR HIGHER, THE ENGINE VIBRATION WILL BE ELIMINATED AND THE RAT INDICATION WILL READ CORRECTLY. THE PILOT SHOULD ACCOMPLISH THIS PROCEDURE PRIOR TO READING RAT TO COMPUTE TAKEOFF N_1 SETTINGS.

If icing conditions are anticipated after takeoff, the tail deice system should be functionally checked prior to dispatch.

The tail deice system is checked by placing the tail deice switch to AUTO. The white **TL DEICE PRESS L** annunciator should illuminate for a few seconds then extinguish for approximately six seconds, then the **TL DEICE PRESS R** annuinciator segment should illuminate for a few seconds and extinguish. After approximately three minutes, the cycle will regenerate. The system may be secured after the second cycle responds properly.

CAUTION

THE TAIL DEICE BOOTS SHOULD NOT BE ACTIVATED AT INDICATED RAT BELOW -40°C (-40°F). BOOT CRACKING MAY RESULT.

Before Takeoff

All anti-ice and deice systems should be on and operating if icing conditions are anticipated during departure. All annunciator lights associated with the anti-ice/deice systems should be out.

CAUTION

LIMIT GROUND OPERATION OF PITOT/STATIC HEAT TO TWO MINUTES TO PRECLUDE DAMAGE TO PITOT STATIC TUBES AND THE ANGLE-OF-ATTACK VANE EXCEPT AS REQUIRED IN ICING CONDITIONS.

Cruise

Check the anti-ice and deice systems for proper operation prior to entering areas in which icing may be encountered. The ignition switches should be selected ON when flying through heavy rain.

Descent

With anti-ice systems on during descent, maintain sufficient power for wing anti-ice. Advance throttles as necessary to extinguish wing anti-ice annunciator lights.

NOTE

Check deice system for proper operation prior to entering areas in which icing may be encountered.

Engine anti-ice is provided at all throttle settings, including idle.

Minimum airspeed for sustained flight in icing (except approach and landing) is 160 KIAS.

Approach and Landing

When reconfiguring for approach and landing (i.e., flaps extended and gear down), and any ice accretion visible on the wing leading edge, regardless of thickness, activate the wing and tail deice system. Continue to monitor the wing leading edge for any reaccumulation.

ABNORMAL OPERATION

WING ANTI-ICE FAILURE (WING ANTI-ICE L OR R CAUTION LIGHT ON)

If a **WING ANTI-ICE L** or **R** annunciator illuminates, normally indicates the bleed air wing inlet temperature is too low. Engine power should be increased to attempt to extinguish the light. The L and/or R ENGINE circuit breaker on the pilot's CB panel in the ANTI-ICE group should be checked IN. The engine anti-ice circuit breakers control both the engine and wing anti-ice systems.

IF WING ANTI-ICE LIGHT REMAINS ON (AFTER TWO MINUTES)

Attempt to deenergize the wing shutoff valve by pulling the respective engine anti-ice circuit breaker to allow bleed-air flow to the wing. If the PRSOV valve opens, a noticeable rise in the respective engine ITT should be observed. Pulling the CB out will not affect operation of the annunciator lights.

The WING XFLOW switch should be placed ON to insure an alternate source of bleed air from the opposite engine is available, especially if a rise in ITT was not observed on the respective engine when the ENGINE anti-ice CB was pulled. If the opposite engine is supplying bleed air to the affected wing after XFLOW was selected, a noticeable ITT should be observed on that engine.

After leaving icing environment, reset the ENGINE anti-ice circuit breaker (if applicable) and select anti-ice switches OFF.

ENGINE ANTI-ICE FAILURE (ENG ANTI-ICE L OR R CAUTION LIGHT ON)

If an **ENG ANTI-ICE L** or **R** annunciator illuminates, may indicate a nacelle inlet or stator vane bleed air valve did not open.

The respective ENGINE anti-ice circuit breaker on the pilot's CB panel should be pulled. Pulling the CB will deenergize the respective engine and wing bleed air valves allowing a flow of bleed air to the engine stators, nacelle lip and the wing. A slight ITT rise should be observed on the corresponding engine if an engine bleed-air valve(s) was stuck closed and opened as the CB was pulled. Pulling the CB out will not affect operation of the annunciator lights.

Monitor the engine inlet or leave icing environment as soon as possible.

After clear of the icing environment, reset the ENGINE anti-ice CB and select the anti-ice switches OFF.

WING BLEED AIR OVERHEAT (WING O'HEAT L OR R CAUTION LIGHT ON)

Light Cycles ON and OFF

Indicates a probable bleed-air leak in the wing heat shield allowing hot bleed air into the ram air cavity impinging on the forward exterior fuel tank enclosure.

If the system operates properly, the PRSOV activates closed as an overtemperature condition is sensed. The **BLD AIR O'HEAT L** or **R** annunciator will illuminate to indicate the overheat condition, but should extinguish after a few seconds after the PRSOV closes. When the annunciator light extinguishes, the PRSOV reopens. Therefore, the annunciator may continue to cycle ON and OFF as the wing overheats and cools down. In this case, the wing anti-ice system may be left ON while in icing conditions.

Attempt to lower the temperature by decreasing power on the respective engine.

Continuous Illumination

Indicates a probable bleed-air leak in the wing heat shield allowing hot bleed air into the ram air cavity impinging on the forward exterior fuel tank enclosure, and the wing PRSOV did not close.

Attempt to lower the temperature by decreasing power on the respective engine.

If Light Does Not Extinguish

Place the affected WING/ENGINE anti-ice switch to ENGINE ON and select WING XFLOW ON. This action should reduce the amount of bleed air into the affected wing, thus effectively reducing temperature. There may be a noticeable split in ITT between the engines.

Leave icing environment as soon as practical.

TAIL DEICE FAILURE (TL DEICE FAIL L OR R CAUTION LIGHT ON)

Increase power as required above 70% N_2 to attempt to increase service air pressure to the boots. Reset the Tail Deice switch OFF, then AUTO or MANUAL.

If TL DEICE FAIL Light Remains On

Leave icing environment as soon as possible.

Before Landing

Flaps are set at TAKEOFF and APPROACH position (15°) during approach and remain at 15° throughout the approach and landing. Observe V_{APP} airspeed minimum during the approach.

NOTE

Multiply charted landing distance by 1.4 for flaps 15° landing.

TAIL DEICE TIMER FAILURE (TL DEICE PRESS L OR R ADVISORY LIGHT FAILS TO ILLUMINATE OR CONTINUES TO CYCLE)

If Advisory Light(s) Fails To Illuminate

Check the TAIL DEICE switch in AUTO position. Check the TAIL DEICE circuit breaker on the pilot's CB panel IN.

If the system is still inoperative, hold the TAIL DEICE switch in MANUAL for a few seconds and repeat at 3 to 5 minute intervals.

If TL DEICE PRESS Advisory Light Remains Illuminated With Switch In OFF Position

Pull the TAIL DEICE CB on the pilot's panel. Reset the circuit breaker as needed to actuate the boots.

Leave the icing environment as soon as practical.

Before Landing

Approach and landing is conducted with flaps set to 15° maximum and observe minimum airspeed, V_{APP} until landing is assured. Multiply charted landing distance by 1.4 for flaps 15°.

WINDSHIELD FAULT (W/S FAULT L OR R CAUTION LIGHT ON)

If W/S O'HEAT L or R Caution Light is on, Refer To Windshield Overheat

On Ground

Correct prior to flight. Indicates the respective controller has detected a fault and shutdown the corresponding windshield anti-ice system. May also be accompanied by a corresponding W/S O'HEAT L or R annunciator caution light.

In Flight

Reset the WINDSHIELD ANTI-ICE switch, OFF then ON.

If W/S FAULT L or R Caution Light Remains Illuminated

Select WINDSHIELD heat on the affected side, OFF.

Leave icing environment as soon as practical.

NOTE

Ice protection will be lost to the outboard and center sections of the affected windshield and the inboard section of the opposite windshield.

WINDSHIELD OVERHEAT (W/S O'HEAT L OR R CAUTION LIGHT ON)

If W/S FAULT L or R and W/S O'HEAT L or R Caution Lights Cycle (Controller Failure, System Cycling on Overtemp Limit)

Turn OFF the affected WINDSHIELD heat switch (unless needed for icing or defog).

Leave icing environment as soon as practical.

NOTE

Ice protection will be lost to the outboard and center sections of the affected windshield and the inboard section of the opposite windshield.

If W/S Fault L or R and W/S O'HEAT L or R Caution Lights on Steady (System Failure)

Indicates overheat protection may be lost. Place the WINDSHIELD heat switch OFF on the affected side.

Leave icing environment as soon as practical.

NOTE

Ice protection will be lost to the outboard and center sections of the affected windshield and the inboard section of the opposite windshield.

COCKPIT FORWARD OR SIDE WINDSHIELD CRACKED OR SHATTERED

Select cabin pressurization to 9,500 feet. If the airplane is above 25,000 feet, MANUAL control will have to be used (cherry picker).

NOTE

Descend to the lowest practical altitude consistent with fuel range requirements; 41,000 feet or lower is recommended.

Either windshield ply is structurally capable of maintaining cabin pressure.

One crew should don an oxygen mask and set regulator to NORMAL. Remain clear of or leave icing conditions.

If Either Forward Windshield Failed

Turn OFF both WINDSHIELD anti-ice switches and land as soon as practical.

If a Side Windshield Failed

Turn OFF the opposite WINDSHIELD anti-ice switch and land as soon as practical.

Pitot-Static Heater Failure (P/S HTR OFF L or R, or STBY P/S HTR Caution Light On)

Check the PITOT & STATIC heat switch ON. Check L and R PITOT STATIC, and STBY P/S HTR circuit breakers IN on pilot's CB panel.

If P/S HTR OFF L or R Annunciator Remains Illuminated

Select autopilot control to cockpit side with operable static heat (FD/AP PFD1/2 selector).

NOTE

The autopilot references the pilot's (L) or copilot's (R) pitot-static system; therefore, the altitude hold and speed hold functions may be inoperative if the coupled side pitot-static system fails in icing conditions. Autopilot control should be transferred to operative side.

If STBY P/S HTR Caution Light Remains Illuminated

If operating in icing conditions, the airspeed and altitude displays on the Meggitt tube may be affected.

ANGLE-OF-ATTACK PROBE HEATER FAILURE (AOA HTR FAIL CAUTION LIGHT ON)

Indicates that the AOA probe heating element has failed. Check the AOA HEATER circuit breaker on the pilot's CB panel IN. If CB checks IN and the annunciator remains illuminated, leave icing environment as soon as practical.

NOTE

If the AOA probe heater fails and the AOA probe becomes iced, the stick shaker may not function and the PFD low airspeed awareness display tape may not be reliable.

LIMITATIONS

Limit ground operation of pitot-static heat to two minutes to preclude damage to pitot tubes and angle-of-attack vane.

Minimum airspeed for sustained flight in icing (except approach and landing) is 160 KIAS.

The airplane is approved for flight into known icing conditions.

Following excerpt from the FAA approved *Citation Excel Airplane Flight Manual* (*AFM*):

"Flight into known icing is the intentional flight into icing conditions that are known to exist by either visual observation or pilot weather report information. Icing conditions exist any time the indicated RAT is +10°C and below, and visible moisture in any form is present. The first visual cue of icing accretion is normally found at the base of the windshield near the center post. The Excel, with properly operating anti-ice and deice equipment, is approved to operate in maximum intermittent and maximum continuous icing conditions as defined by FAR 25, Appendix C. The equipment has not been designed to provide protection against freezing rain or severe conditions of mixed or clear ice. During all operations, the pilot is expected to exercise good judgement and be prepared to alter the flight plan, i.e., exit icing, if conditions exceed the capability of the aircraft and equipment.

Ice accumulations significantly alter the shape of airfoils and increase the weight of the aircraft. Flight with ice accumulated on the aircraft will increase stall speeds and alter the speeds for optimum performance. Flight at high angle-of-attack (low airspeed) can result in ice building on the underside of the wings and the horizontal tail aft of areas protected by boots or leading edge anti-ice systems. Minimum airspeed for sustained flight in icing conditions (except approach and landing) is 160 KIAS. Prolonged flight with flaps and/or landing gear down is not recommended. Trace or light amounts of icing on the horizontal tail can significantly alter airfoil characteristics which will affect stability and control of the aircraft.

Freezing rain and clear ice will be deposited in layers over the entire surface of the airplane and can "run back" over control surfaces before freezing. Rime ice is an opaque, granular and rough deposit of ice that usually forms on the leading edges of wings, tail surfaces, pylons, engine inlets, antennas, etc."

QUESTIONS

1. Ice detection is accomplished by:
   A. Visual indications.
   B. ICE DETECT annunciator light.
   C. Wing ice-detector sensor.
   D. Both B and C.

2. P_3 air leaving the precooler too hot, illuminates the:
   A. **ACM O'HEAT** annunciator.
   B. **BLD AIR O'HEAT** annunciator.
   C. **AIR DUCT O'HEAT** annunciator.
   D. **PRECLR O'HEAT** annunciator.

3. Ground cooling of precooled bleed air is accomplished by:
   A. Tailcone ambient air.
   B. The Vapor Cycle Air Conditioner.
   C. Ambient or engine bypass air.
   D. Engine $P_{2.8}$ air.

4. In-flight cooling of bleed air through the precooler is accomplished by:
   A. Service air.
   B. Engine bypass air.
   C $P_{2.8}$ bleed air.
   D. Ram airflow.

5. Bleed air flowing into the wing leading edge panels that is too cold is annunciated by:
   A. **WING TOO CLD L/R** annunciator.
   B. **BLD AIR O'HEA**T annunciator.
   C. No annunciation.
   D. **WING ANTI-ICE L/R** annunciator.

6. The **W/S O'HEAT L/R** annunciator illuminates:
   A. The system must be shutdown immediately.
   B. The **W/S FAULT L/R** annunciator will also illuminate.
   C. Advisory, no action required.
   D. Indicates that the respective forward side window is overheated.

7. If the **TL DEICE FAIL L/R** annunciator illuminates:
   A. In MANUAL mode, considered normal.
   B. In MANUAL mode, the timer is inoperative.
   C. In AUTO mode, the inflation pressure may be too low.
   D. In MANUAL mode, boot did not deflate.

CHAPTER 11
AIR CONDITIONING

CONTENTS

ILLUSTRATIONS

CHAPTER 11
AIR CONDITIONING

INTRODUCTION

The Citation EXCEL air conditioning system provides conditioned air to both the cockpit and cabin. Engine (P_3) bleed air or bleed air from the optional Auxiliary Power Unit (APU), supplies air required to operate the system. Cabin and cockpit temperature is regulated by mixing hot bleed air with air cooled by an Air Cycle Machine (ACM). Various fans are provided to help circulate cockpit and cabin air. A vapor cycle refrigerant type air conditioning unit installed in the tailcone provides supplemental cooling as required, (not available if an optional APU is installed).

GENERAL

The flight crew is provided with automatic and manual temperature controls to regulate the cabin and cockpit environments separately. High Pressure P_3 bleed air is normally tapped off each engine, routed through the engine pylon precoolers and then directed into the tailcone for distribution to the Environmental Control Unit (ECU). Bleed air is further cooled through the ECU and then ducted to the cockpit and cabin outlets. There are three air distribution networks: The overhead cold air ducts, the lower cabin air ductwork, and the cockpit air duct system.

Temperature is normally controlled by mixing hot bleed air (that bypasses the ACM), with a portion of the cold air processed by the ACM. This air mixture is then directed through the underfloor ducting to both the cockpit and cabin. The remainder of the cold air supplied by the ACM is distributed directly to the overhead ducts throughout the cabin and the cockpit. The cabin and cockpit may be supplied with fresh ambient air if the airplane is flown unpressurized.

SYSTEM DESCRIPTION

AIR CONDITIONING

General

Bleed air from the engines passes through precoolers mounted in the engine pylons (refer to Chapter 9, PNEUMATICS). Bleed air is conditionally cooled in the precoolers and then sent through ozone converters (refer to Chapter 9, PNEUMATICS) which enhances air quality. Air is then directed to the ECU/ACM.

ECU Description

The Environmental Control Unit (ECU) utilizes bleed air from the engines or the APU for operation and provides cooling, heating, and pressurization for the cockpit and cabin. The ECU consists of a primary heat exchanger, secondary heat exchanger, an Air Cycle Machine (ACM consisting of a compressor, turbine and fan), a water separator, and an over temperature switch (Figure 11-1).

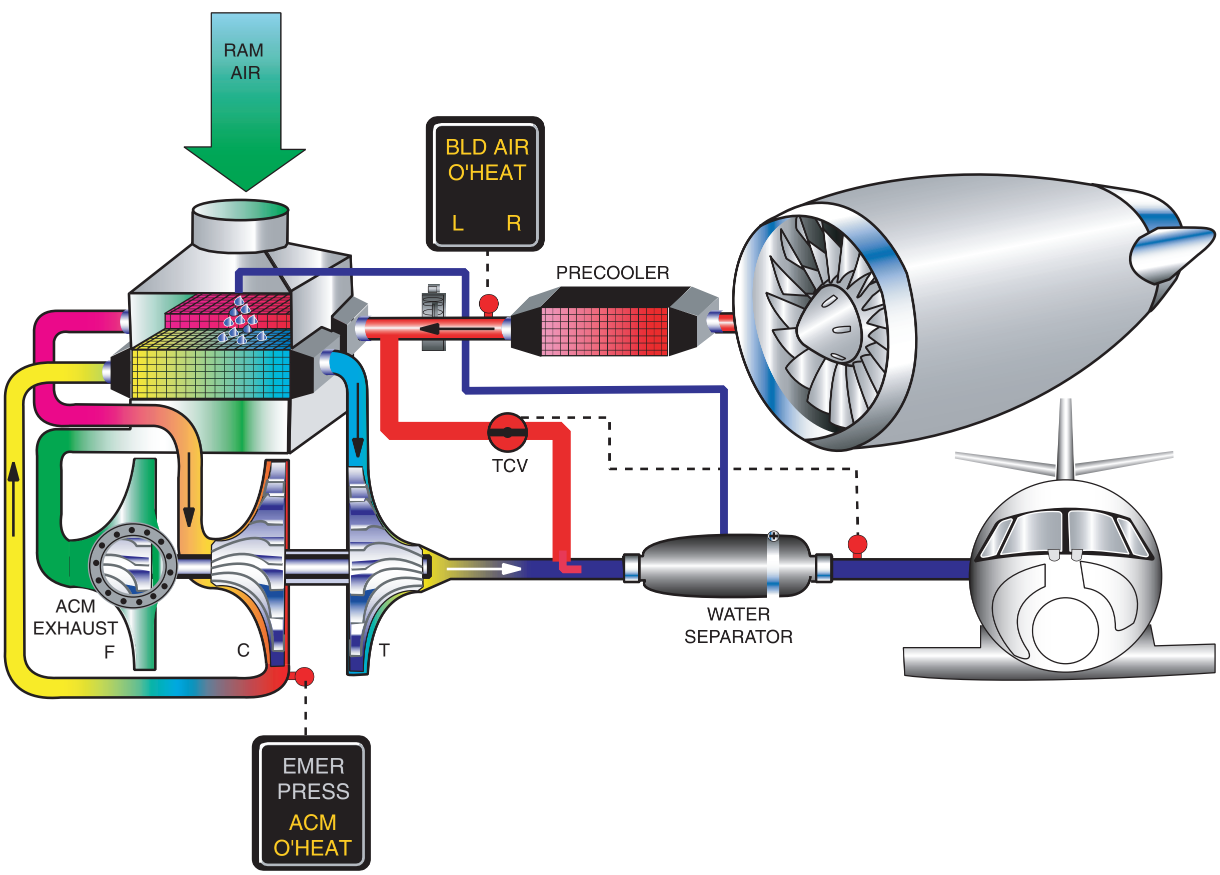


Figure 11-1 Environmental Control Unit (ECU)

ECU Control

A five-position PRESS SOURCE selector switch (Figure 11-2) located on the tilt panel, is used to select bleed air sources to the ECU from either or both engines. The PRESS SOURCE selector switch has positions labeled “OFF-LH-NORM-RH-EMER.”

Figure 11-2 PRESS SOURCE Selector

The OFF position closes all environmental bleed-air valves. The LH and RH normal flow control valves are energized closed by main DC power (Figure 11-3).

Selecting LH or RH (single source), de-energizes the selected flow control valve open and energizes the opposite valve closed.

Selecting NORM de-energizes both left and right normal flow control valves open allowing both engines to provide bleed air to the ECU (Figure 11-3).

Selecting EMER energizes the EMER valve open and bleed air from the left engine precooler flows directly into the cabin. EMER position also electrically closes both normal flow control valves, shutting off air flow through the ECU (Figure 11-3).

Air Cycle Machine (ACM)

Bleed air enters the ACM compressor from the primary heat exchanger, where it is compressed and routed to the secondary heat exchanger. From the secondary heat exchanger, the cooled compressed air drives the turbine wheel. The air rapidly expands as it leaves the turbine wheel producing cold air. Turbine power is used to drive the compressor and the cooling fan which draws air across the heat exchangers (Figure 11-1).

The ACM compressor, turbine and fan assembly are all mounted on a common shaft and supported by air bearings. They ride on a film of compressed air as the ACM begins to rotate.

Primary Heat Exchangers

The primary and secondary heat exchangers are joined as a pair and arranged in parallel for ram air flow (Figure 11-1).

NACA scoops located on the dorsal fin, supply ram air for the heat exchangers (Figure 11-4). With the airplane on the ground, a fan connected to the ACM turbine shaft draws air through the dorsal fin air inlets, through the heat exchangers and pumps it overboard through a louvered duct on the right side of the tailcone (Figure 11-5). The fan inlet pressure is boosted in flight by ram air pressure.

Due to the large heat increase as the bleed air is compressed within the ACM, the heat exchangers assist in cooling efficiency.

Water Separator

The water separator (Figure 11-6) is comprised of a coalescer, to collect moisture, and a bypass valve, to allow air to bypass the water separator if the coalescer should freeze.

Cool air from the ACM is ducted to the water separator. Moisture collected by the coalescer is routed via drain tubes to the water aspirator located on the ACM inlet duct and is ejected onto the secondary heat exchanger (Figure11-1). Dehumidified cool air from the water separator is distributed to the cabin inlet ducts.

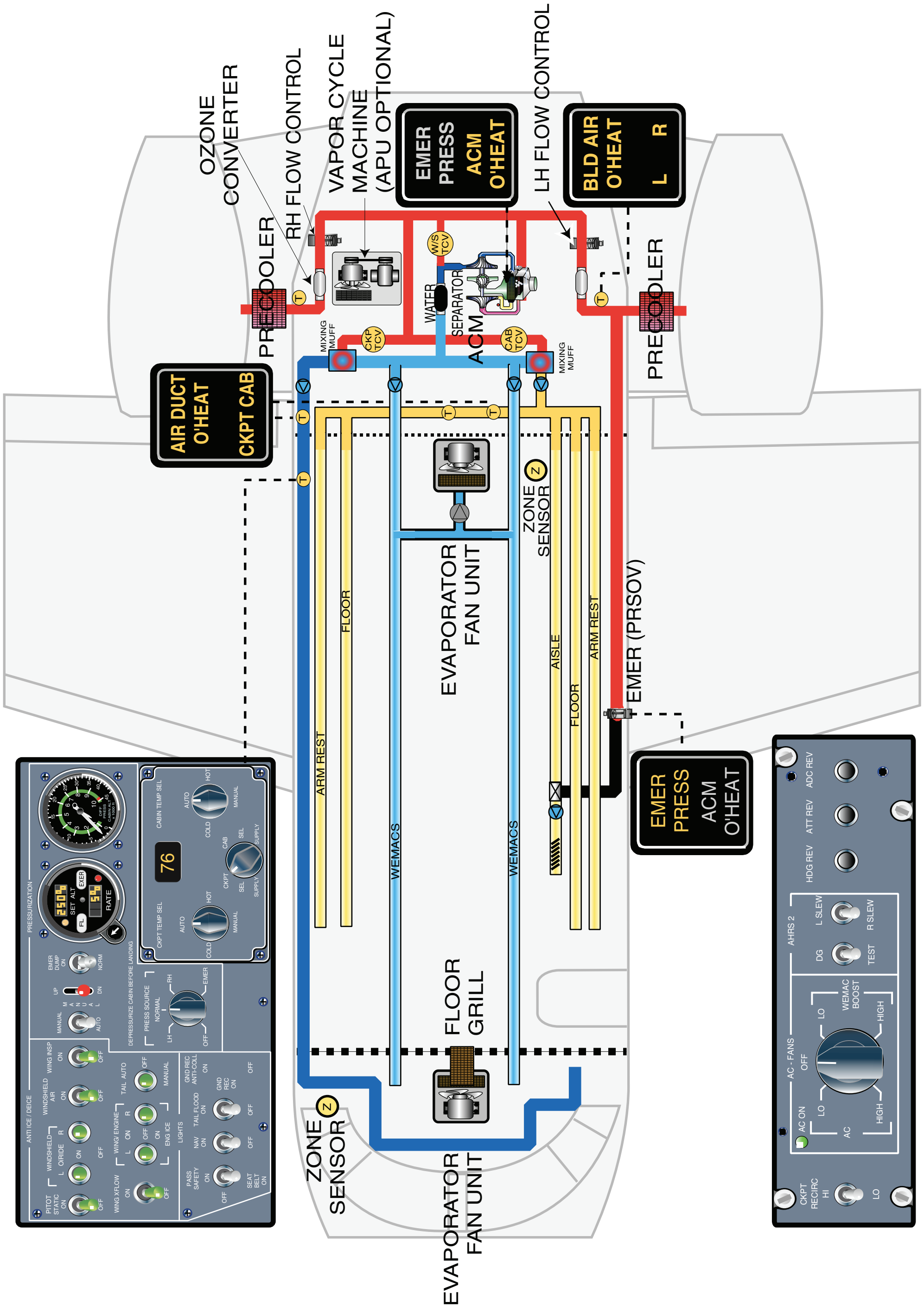


Figure 11-3 Air Conditioning Schematic

Figure 11-4 Dorsal Fin Ram Air Inlet

Figure 11-5 ACM Exhaust

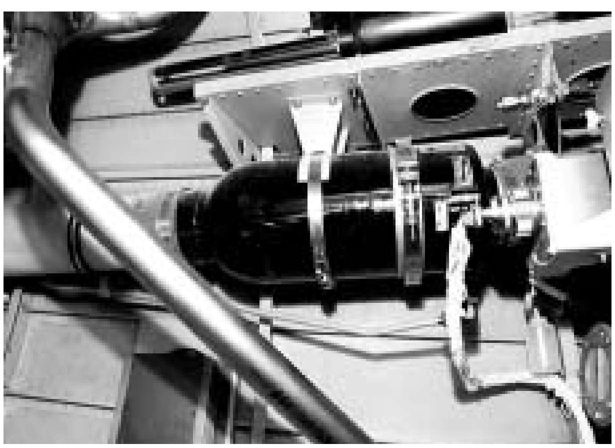


Figure 11-6 Water Separator

Low Temperature Control

The 35°F temperature control system consists of a temperature sensor and a temperature control valve (TCV). The temperature sensor located downstream of the water separator monitors the cool air temperature from the water separator. If temperature falls below 35°F, a signal is sent to the temperature control valve located in the bleed air inlet duct (Figure 11-3), and causes the valve to modulate toward open. As the TCV modulates open, a variable amount of hot bleed air is introduced into the ACM outlet to mix with cold air to prevent water from freezing in the water separator. This mixing of hot air with cool air from the ACM ensures air temperature from the water separator will not fall below 35°F.

A restrictive orifice downstream from the TCV limits the amount of hot bleed air that can be mixed with conditioned cool air in the event a temperature control failure is experienced, i.e., TCV stuck open.

ACM Overtemperature Protection

An overtemperature switch is located in the ACM compressor discharge outlet duct (Figure 11-1). The switch monitors bleed air temperature exiting the ACM compressor, and is designed to protect the ACM from excessively high temperature.

If compressor discharge temperature exceeds approximately 420°F, a logic module will cause the left and right flow control valves to "close," shutting off all bleed air to the ACM. The overtemperature sensor will cause the **ACM O'HEAT** annunciator to illuminate "flashing." In flight, if the sensor detects an overheat condition, the logic module will also trip the emergency pressurization valve "open," to prevent the cabin from losing pressure at altitude. The emergency valve trips "open" simultaneously as the normal flow control valves close.

The **EMER PRESS** annunciator light will illuminate "flashing" simultaneously with the **ACM O'HEAT** annunciator. When the temperature drops to an acceptable level, the logic module deenergizes the left and right flow control valves open and the emergency pressurization valve closed (the **ACM O'HEAT** and **EMER PRESS** lights extinguish), allowing the system to return to normal operation.

ECU Operation

NORM — With the PRESS SOURCE selector switch in NORM, precooled engine bleed air (approximately 475°F) enters the ACM from the two, LH and RH, normal flow control valves. The valves decrease bleed air pressure to control total air volume entering the ECU at approximately 20 lbs/min at sea level and 12 lbs/min at FL 450. Bleed air enters the ACM at the primary heat exchanger and cooled to approximately 200-300°F prior to entering the ACM compressor. The compression process raises the temperature to approximately 300-400°F. Bleed air then exits the ACM and enters the secondary heat exchanger and cooled to approximately 100-150°F prior to entry into the ACM turbine wheel. The expansion process through the turbine assembly cools the air to approximately 40-50ºF on a hot day. On moderately cold days, the turbine outlet temperatures will drop well below freezing. To prevent ice particles from freezing in the water separator and blocking airflow, cold turbine air is mixed with hot bleed air that bypasses the ACM (discussed previously in this section).

SINGLE SOURCE — Positioning the PRESS SOURCE selector to single-source, LH or RH, reduces total airflow through the ECU by approximately one-half. However, the outflow valves will compensate (move toward closed) and prevent cabin differential pressure from decreasing (refer to Chapter 12, PRESSURIZATION).

OFF — Selecting the PRESS SOURCE selector OFF, will shut off all bleed air to the ECU and cause the cabin to begin depressurizing. The system requires main DC power to close the normal flow control valves. As cabin altitude climbs to approximately 14,500 feet, the emergency pressurization valve automatically opens (**EMER PRESS** annunciator "flashes") and prevents the cabin from exceeding 14,500 feet. The red **CAB ALT** annunciator will illuminate "flashing" as the cabin altitude passes 10,000 feet or 14,500 feet (pressurization system operating in the high altitude mode, refer to Chapter 12, PRESSURIZATION).

NOTE

> Selecting the PRESS SOURCE selector OFF, will cause the cabin altititude to climb at approximately 500-600 feet per minute (normal cabin leak rate).

If a loss of main DC power occurs, both the LH and RH flow control valves deenergize to the "open-fail safe" position and prevent the cabin from depressurizing.

EMER — Placing the PRESS SOURCE selector to EMER, electrically opens the emergency pressurization valve (**EMER PRESS** annunciator illuminates "flashing"), and electrically closes both LH and RH flow control valves (Figure 11-3). Emergency pressurization, routes bleed air from the left engine precooler directly to the forward cabin dropped aisle duct. EMER Bleed air passes through a venturi in the underfloor network, which draws cooler ambient cabin air to mix with hot bleed air to moderate the temperature entering the cabin.

Malfunctions that may require the crew to select EMER PRESS are: Rapid loss of cabin pressure resulting from, failure of the ACM, duct failure from the ECU to the aft pressure bulkhead, fuselage structural failure, etc.

Selecting EMER on the ground, will cause the **EMER PRESS** annunciator to illuminate, but the emergency pressurization valve will remain closed (LH squat switch).

AIR DISTRIBUTION

GENERAL

The cabin and cockpit air distribution systems direct flow of conditioned air to provide a comfortable ventilated cabin and cockpit through three separate networks: (1) Overhead cold air ducting; (2) lower cabin; and (3) cockpit conditioned air networks.

Approximately one-half of the ACM output is distributed directly through the cabin/cockpit overhead network. The other half is mixed with hot bleed air via mixer assemblies to produce conditioned air through the underfloor ducting to the cockpit and cabin.

The Vapor Cycle Air Conditioner system provides supplemental cooling to the cabin and cockpit ventilation system through forward and aft evaporator/blower assemblies. Various outlets and fans help circulate air throughout the cabin and cockpit.

The ECU temperature control panel is located on the center tilt panel (Figure 11-7).

OVERHEAD COLD AIR DISTRIBUTION

The overhead cold air distribution system originates from the ACM in the tailcone. At the aft pressure bulkhead, each left and right overhead duct is connected to a check valve to prevent reverse flow. The formed duct work travels from the check valves forward through the cabin overhead above the passenger seats and terminates on both the left and right sides near the cabin entry door (Figure 11-3). Flexible ducting is connected to the forward end of the formed cabin ducts to extend the network into the cockpit terminating above the crew seats.

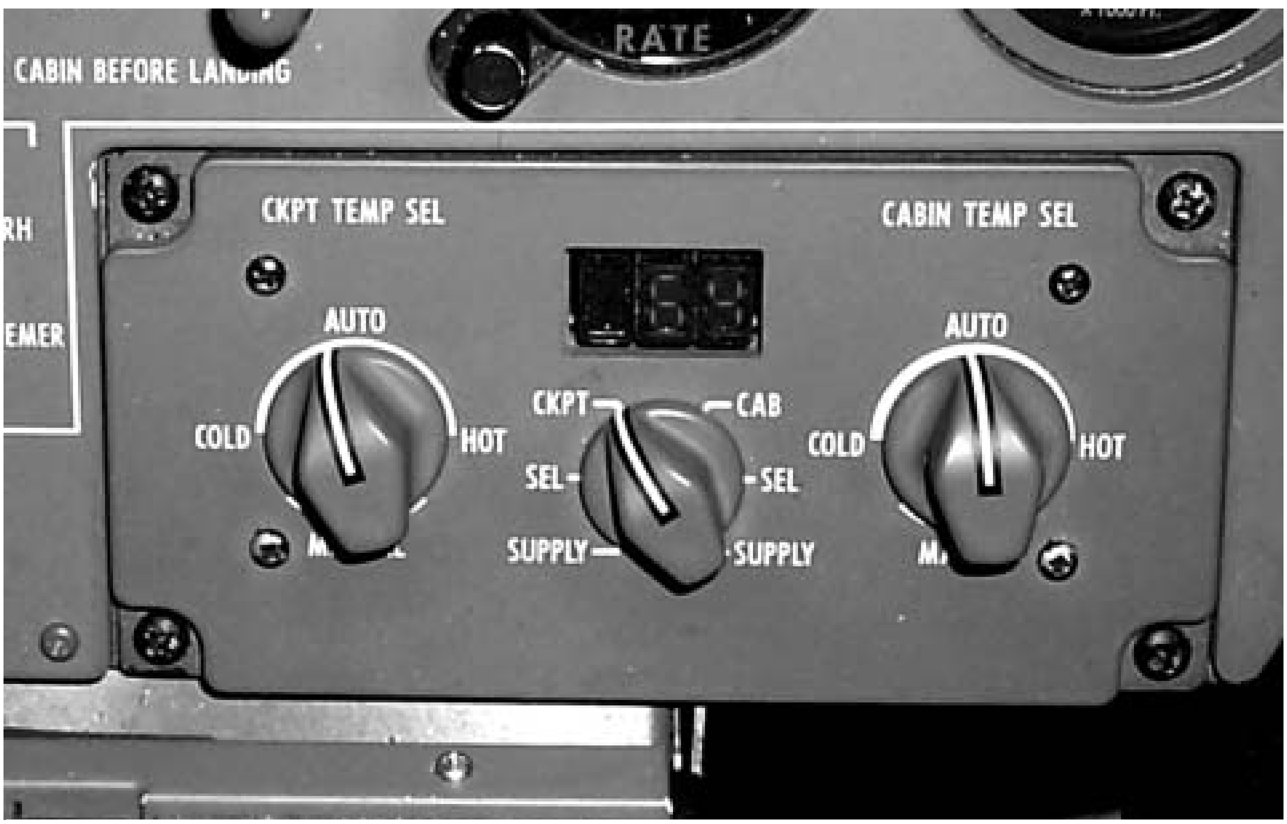


Figure 11-7 Environmental Control Panel (ECU)

The overhead ductwork distributes cold air only to the cabin and cockpit from the ACM outlet in the tailcone. Wemac outlets are installed in the duct work above each passenger seat, and above each crew seat (Figures 11-8 and 11-9). The Wemac outlets are operated from full open to full closed individually as desired by each cabin occupant and each crew member.

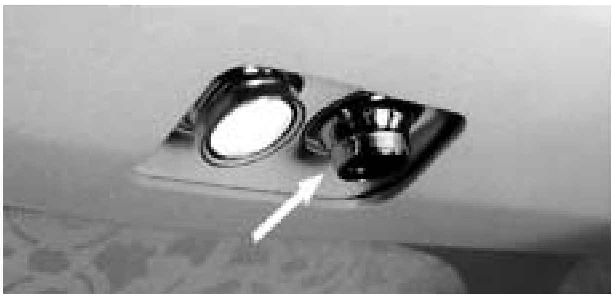


Figure 11-8 Cabin Wemac Vent

Figure 11-9 Cockpit Wemac Vents

WARM AIR DISTRIBUTION

Warm air is created when cool air from the acm is mixed with hot bleed air through the "mixing muffs" to modulate air temperature (Figure 11-3). This mixing takes place in the tailcone area. The warm air is then routed to the cabin and cockpit via a series of ducts, hoses and valves.

CABIN AIR DISTRIBUTION

General

From a check valve located on the aft pressure bulkhead, warm air is routed to both sides of the cabin. On the left side, warm air is routed underneath the floorboards for dropped aisle heating, and through side wall ducting to integral foot warmers and arm rest diffusers. On the right side, warm air is routed through side wall ducting to integral foot warmers and arm rest diffusers only.

Lower Cabin Air Distribution

The lower cabin air distribution system is supplied air through the LH lower supply duct. As air enters the cabin network, it splits into several paths; The LH armrest and footwarmer ducts, the dropped aisle ducts along both sides of the aisle, and to the RH armrest and footwarmer ducts. The footwarmer and armrest ducts are piccolo tubes that allows the air to flow evenly over the length of the cabin. The ducting design allows warm air to discharge into the cabin environment.

The left side ducting extends from the aft cabin forward to the main entrance door. The right side ducting extends from the aft cabin forward to just aft of the cabin/cockpit divider. A wyed crossover plenum on the forward side of the aft pressure bulkhead connects the LH and RH sidewall ducts and the dropped aisle ducts together.

COCKPIT AIR DISTRIBUTION

General

The cockpit air distribution system is supplied with conditioned air through the RH lower supply duct (Figure 11-3). Air enters the aft cabin through a check valve and ducted underneath the RH seats forward to the cockpit. In the cockpit, air is split off into sidewall diffusers and forward bulkhead footwarmer diffusers.

Cockpit Vent System

The cockpit conditioned air vent system consists of side console air outlets, footwarmers and instrument panel console Wemacs. The side console air outlets are small holes drilled in the top surface of the pilot and copilot's side consoles, directly below the side windows (Figure 11-10). The vents are open and closed with "WINDOW" knobs located on the side walls adjacent to each side of the instrument panel directly in front of the pilot's and copilot's circuit breaker panels (Figure 11-11). The LH and RH side consoles vents are interconnected through a crossover duct that extends from the right sidewall to the left side wall.

The LH and RH footwarmer outlets are mounted in the lower forward crossover duct and are opened and closed by knobs located directly below the side console vent knobs adjacent and forward of each circuit breaker panel (Figure 11-11).

Mounted on either side of the instrument panel are forward cockpit Wemac vents that may be rotated open and closed as desired (Figure 11-12). Air is supplied to these vents from flex ducts connected to a center outlet between the LH and RH footwarmers on the lower forward crossover duct.

A console Wemac fan is located in the center of the forward footwarmer crossover duct. The fan is activated electrically by a CPT RECIRC HI–OFF–LO switch on the copilot's lower instrument panel adjacent to the vapor cycle control panel (Figure 11-13). Operating the fan knob in either the HI or LO position, will generate air flow through both forward cockpit wemac vents.

Figure 11-10 Side Console Vent

Figure 11-12 Forward Cockpit Wemac Vents

Figure 11-11 Side Console Vent Knobs

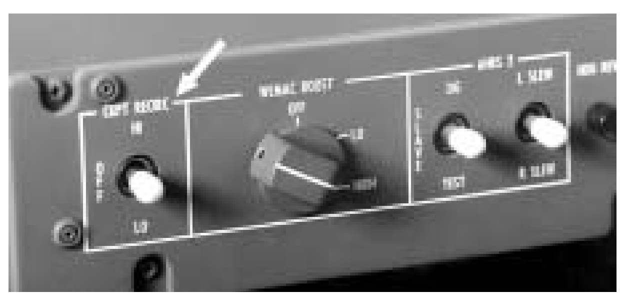


Figure 11-13 CKPT Fan Switch APU Installation

The cockpit forward side windows are electrically heated to prevent fog and frost from accumulating (refer to Chapter 10, ICE AND RAIN PROTECTION). To prevent condensation from accumulating on the inside of the cockpit side windows during high humidity conditions, open the side console "WINDOW" vent knobs and increase the cockpit temperature slightly.

TEMPERATURE CONTROL

GENERAL

Cockpit and cabin conditioned air temperatures are controlled individually from the cockpit tilt panel (Figure 11-7).

Temperature is controlled in the cabin and cockpit by mixing constant temperature cool air from the ECU water separator (approximately 35°F) with unconditioned warm bleed air (Figure 11-3). Controls and valves are designed to vary temperature in the warm air distribution network as cabin and cockpit environmental conditions change in order to maintain constant temperatures, i.e., aircraft climbing and descending.

Components which are common to both the cabin and cockpit temperature control systems include Temperature Control Valves (TCV), mixing muffs, temperature sensors, zone sensors, and duct overheat switches. The temperature controller is part of the cockpit ECU panel (Figure 11-7), and provides the flight crew with the ability to control temperatures separately for the cockpit and cabin.

COMPONENTS

Temperature Control Valves (TCV) and Mixing Muffs

Three temperature control valves (TCV) are used to regulate the amount of hot bleed air mixed with cold air from the ECU. The valves are butterfly type valves that are operated by brushless DC motors. The DC motor of each valve receives pulses of power from the controller to position the butterfly.

One is located between the bleed air duct and the ACM outlet, and modulates open as required, to allow a variable amount of hot air to mix with cold ACM air to protect the water separator from freezing (previously discussed in this chapter). The low temperature sensor, located in the water separator outlet, is dedicated to the water separator TCV.

Two TCV's are located just aft of the aft pressure bulkhead. As pulses of power are received from the controller, the TCV motor modulates the butterfly valve, varying the amount of hot bleed air that is mixed with cool ACM air, resulting in controlled cabin and cockpit temperatures as it enters the respective warm air distribution ducts. Hot bleed air from the TCV is mixed with cool ACM air through the mixing muffs (Figure 11-3).

The mixing muffs surrounds the conditioned air ducts and inject hot bleed air into the conditioned cold air ducts. The mixed air becomes temperature controlled and is routed to the cabin and the cockpit via dedicated lower cabin air ducts.

The LH TCV is connected to the cabin warm air distribution system, and the RH TCV is connected to the cockpit warm air distribution system (Figure 11-3).

Duct Supply Temperature Sensors

Two duct temperature sensors are located under the cabin floorboards just downstream of the aft pressure vessel check valves in the cabin and cockpit distribution ducts. Each sensor monitors the temperature of warm air entering the ducts and sends this information to the temperature controller.

Duct Overheat Switches

Duct overheat switches are mounted adjacent to each duct temperature sensor. These switches activate the respective **AIR DUCT O'HEAT CKPT** and/or **CAB** annunciator "flashing" if temperature in the respective duct exceeds 300°F.

Zone Temperature Sensors

Two zone temperature sensors are incorporated in the system, one mounted in the cockpit RH side console, and one mounted in the aft cabin area in the Passenger Service Unit (PSU) above the left aft seat. The sensors are mounted in a box with a small fan that pulls a sample of cabin/cockpit air across the respective sensor to provide a representative zone temperature. The sensors monitor the temperature of the air in the cockpit and cabin, and provide a reference temperature to the temperature controller.

Temperature Controller

A single temperature controller is located on the tilt panel directly below the pressurization controller (Figure 11-7). All logic for the temperature control system is contained in the temperature controller. The controller receives input signals from the zone sensors and duct temperature sensors. It in turn, sends a DC current signal to the temperature control valves (TCV) to modulate toward open or closed as required to attain or maintain desired temperature .

Selector-Indicator

A single temperature control selector-indicator is integrated into the temperature controller. The selector-indicator consists of cabin and cockpit temperature selectors, a digital temperature indicator and a display selector. The digital temperature indicator window, displays temperature in degrees Fahrenheit.

The crew can select desired temperature for the cockpit and cabin by rotating the CABIN TEMP SEL and the CKPT TEMP SEL rotary selectors while pointing toward the top 180° AUTO arc, provided the lower center knob is in the respective CKPT–SEL or CAB–SEL position. The range of temperature selection in AUTO, is a minimum of 65°F (COLD) to a maximum of 85°F (HOT) for both the cabin and cockpit. Selecting 12 o'clock, AUTO position with either selector, the indicator should display approximately 75°F.

Positioning the CKPT TEMP SEL and/or the CABIN TEMP SEL knobs in MANUAL (bottom 180° arc), and releasing the knob(s), the knob(s) spring load to a 6 o'clock detent. Operating in MANUAL, the respective temperature control valve (TCV) for the cabin and cockpit, located in the tailcone, are directly controlled by the respective selector (open or closed as desired). Moving the desired selector, CKPT or CABIN, out of the detented 6 o'clock position and holding it toward the HOT or COLD position, electrically drives the corresponding TCV butterfly valve toward open (hot) or toward closed (cold). TCV valve travel time in MANUAL from full open (hot) to full closed (cold), or vice versa, is approximately 10 seconds.

NOTE

While operating in MANUAL with the center selector in cockpit or cabin SEL, any displayed temperature indications are meaningless.

Positioning the center selector switch to CKPT or CAB, displays the current temperature recorded by the respective zone temperature sensor. Moving the switch to SUPPLY–CKPT or SUPPLY–CAB, the temperature display is from the respective duct temperature sensor located in the corresponding rear supply duct (cockpit or cabin). The supply duct temperature may indicate much warmer or colder than the corresponding zone temperature indications.

SUPPLEMENTAL AIR CONDITIONING — VAPOR CYCLE SYSTEM

NOTE

If an optional APU is installed, the vapor cycle system will be removed.

General

Additional supplemental cooling of the cabin and cockpit is provided by a Vapor Cycle air conditioning unit mounted in the tailcone. It is a conventional compressor/ condenser R134a system powered from the main DC electrical system. It may be operated by itself or in conjunction with the ECU. It is normally operated in conjunction with the ECU to provide additional cooling during warm environmental conditions, both on the ground or inflight.

An electric motor drives the vapor cycle cooling system compressor which pumps refrigerant through the system. The hot gaseous refrigerant from the compressor is condensed to a liquid by airflow through the condenser. The cooled liquefied refrigerant is expanded to a low temperature gas by expansion valves at each evaporator. The cold gas in the evaporators removes heat from the cabin air as it is circulated through the evaporators by evaporator blower fans

The system consists of two cabin evaporators (one forward and one aft), a tailcone mounted condenser, compressor and motor (Figure 11-14), and a control unit located on the copilot's lower instrument panel (Figure 11-15).

Compressor

The belt driven rotary piston-type compressor is mounted on a pallet with the electrical drive motor and condenser. The pallet is located in the upper forward right side of the tailcone (Figure 11-14).

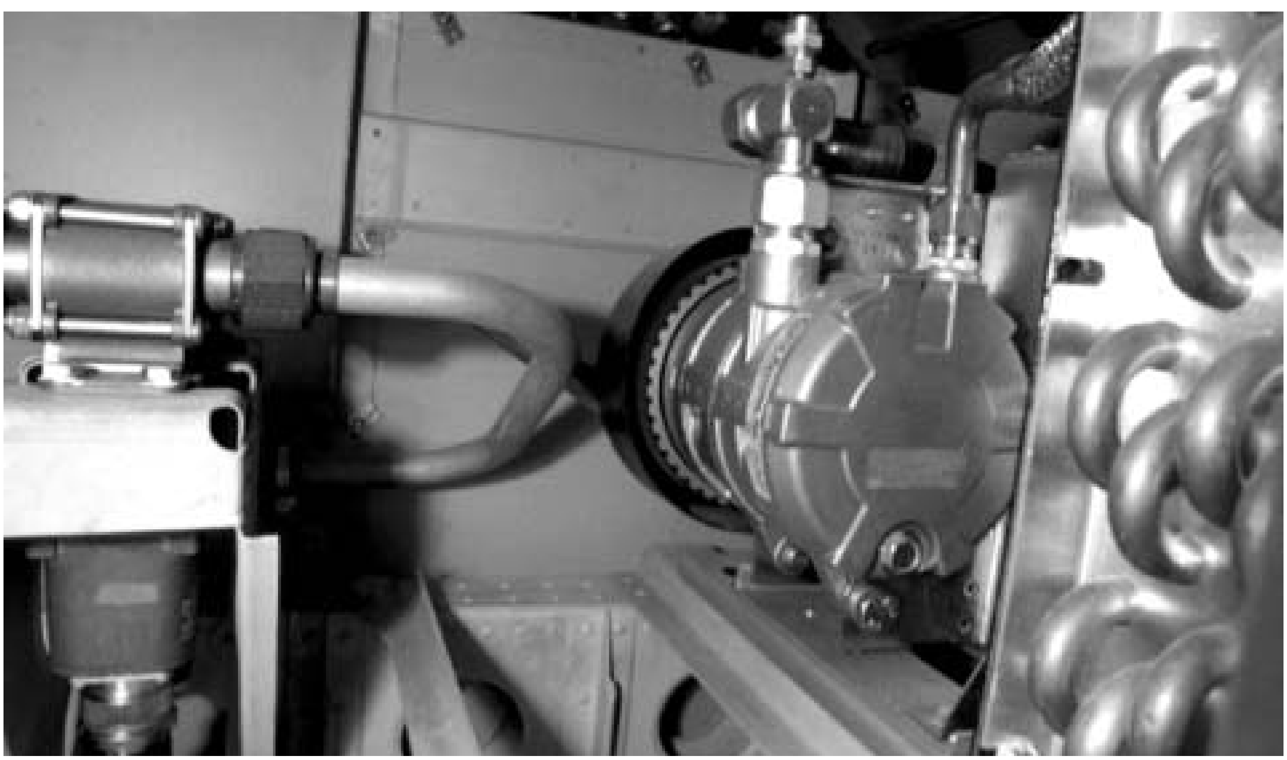


Figure 11-14 Vapor Cycle Air Conditioner, Tailcone

Figure 11-15 Vapor Cycle A/C Control Panel

Condenser

The condenser is mounted on the pallet aft of the compressor and motor (Figure 11-14). The condenser ram air inlet and exhaust outlet are ducted through the tailcone right sidewall skin (Figure 11-16). A fan drives air through the condenser and out the exhaust duct. A receiver/dryer associated with the condenser removes moisture from the refrigerant. The receiver/dryer also serves as a reservoir to separate the liquid from the gaseous refrigerant, allowing only liquid refrigerant to continue cycling.

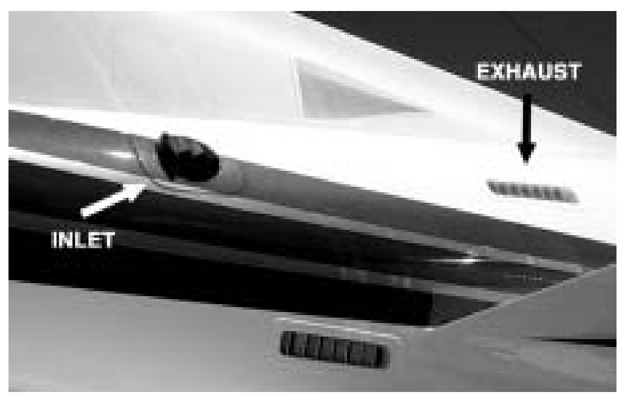


Figure 11-16 Vapor Cycle Inlet & Exhaust

Evaporator/Blowers

Two evaporator/blowers are installed in the cabin. The forward evaporator is mounted in the floor of the forward end of the dropped aisle just prior to entry into the cockpit. The aft evaporator is connected to the overhead distribution system (Figure 11-17). The cold air is driven across the evaporator coils by electrically powered centrifugal blowers.

The forward evaporator/blower mounted in the forward end of the cabin dropped aisle floor just prior to entry into the cockpit, is designed to blow cold air upward through a

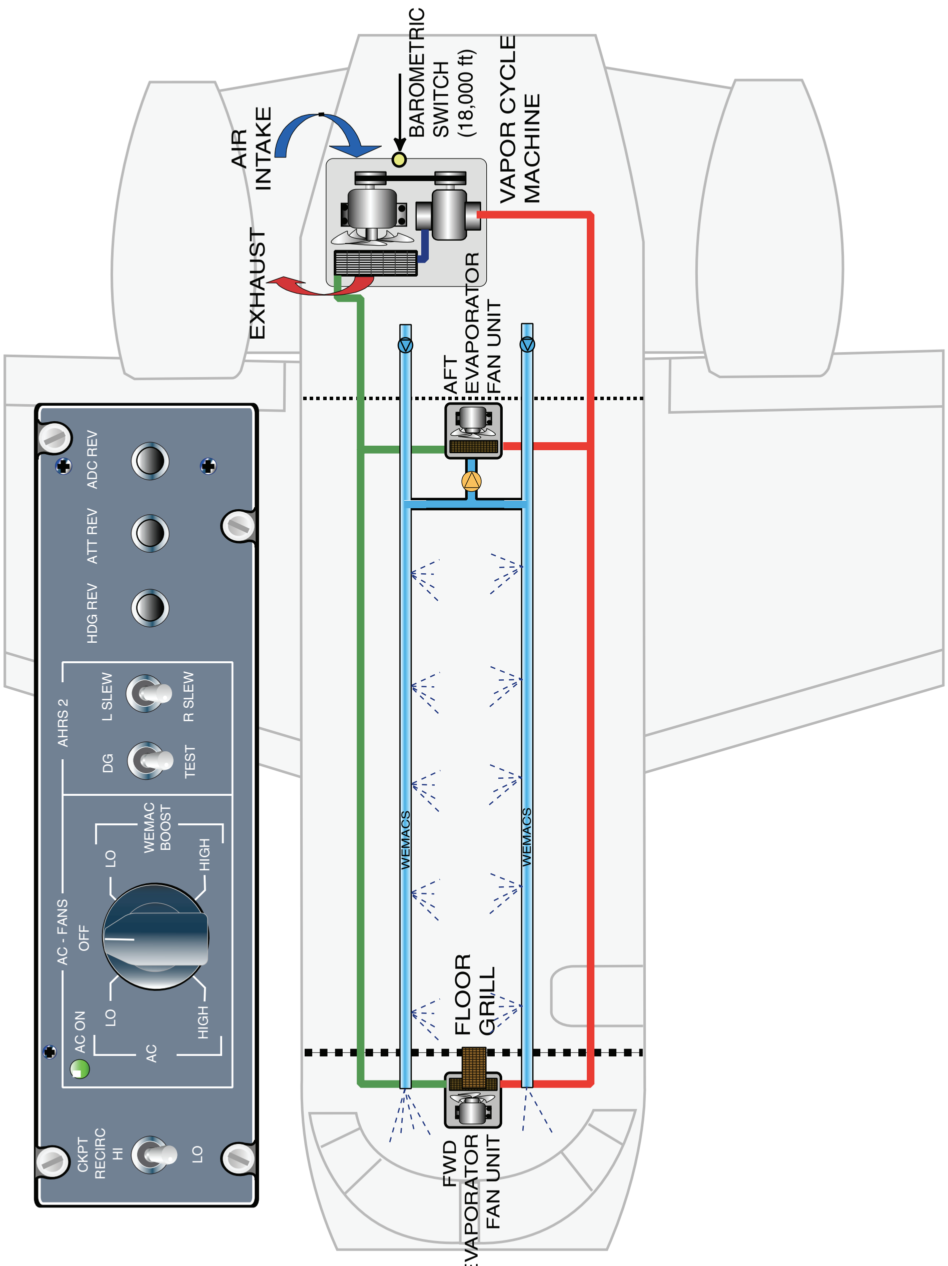


Figure 11-17 Vapor Cycle Air Conditioning System

grille that directs air flow both forward toward the cockpit and aft into the cabin.

The aft evaporator/blower is mounted in the vanity, just forward of the aft pressure bulkhead. The aft evaporator/blower assembly is connected to the overhead cold air distribution system and its air is distributed through the cabin and cockpit through the overhead wemac outlets.

Operation

Controls for the vapor cycle system are located on the copilot's lower instrument panel (Figure 11-15).

A single rotary control switch on the panel controls the system The center OFF position, deenergizes the system completely.

Placing the switch to the left, AC-HI or LO, activates the compressor motor and the evaporator blower fans. A green indicator light labeled AC/ON illuminates to indicate compressor operation (Figure 11-15). With the switch in HI (left), both evaporator fans operate at high speed, and positioned to LO (left) they operate at low speed.

The system is designed to operate on the ground and in flight up to 18,000 feet.

NOTE

> In flight, both generators must be operating in order for the compressor drive motor to operate. If one generator fails, the drive motor will automatically disconnect from its power source.

Wemac Boost

The HI-WEMAC BOOST-LO position of the rotary switch, controls only the aft evaporator fan either high or low speed (Figure 11-17). The compressor and the forward evaporator fan do not operate with the switch positioned to the right (AC/ON indicator light extinguished).

Operating in the WEMAC BOOST HI or LO position, the aft fan draws cabin air through the non-operating aft evaporator and forces the air through both overhead cool air distribution ducts, creating greater air flow through all Wemac outlets. An in-line flapper-type check valve in the Wemac line, prevents reverse flow of cool air when the aft evaporator fan is not operating.

There are no restrictions associated with operating the Wemac Boost system. It may be operated on the ground or in flight at all altitudes and temperatures if desired.

NOTE

> Aircraft with APU installations will have the WEMAC BOOST LO – HIGH switch only (Figure 11-13). All reference to the Vapor Cycle Air Conditioner will be removed.

Protection

A barometric switch is incorporated to shutdown the compressor drive motor above 18,000 feet (Figure 11-18). The compressor will shutdown automatically climbing through 18,000 feet as indicated by the green indicator light extinguishing on the control panel. Adjacent to the barometric switch, is an hour meter to record operating time on the system (Figure 11-19).

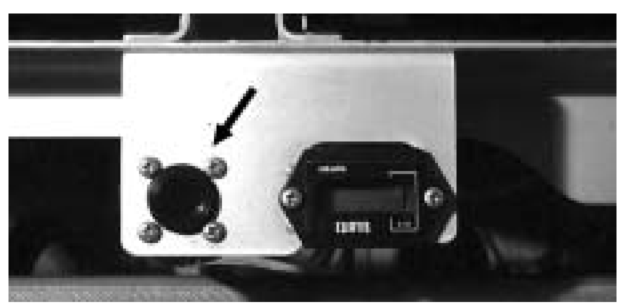


Figure 11-18 Barometric Switch

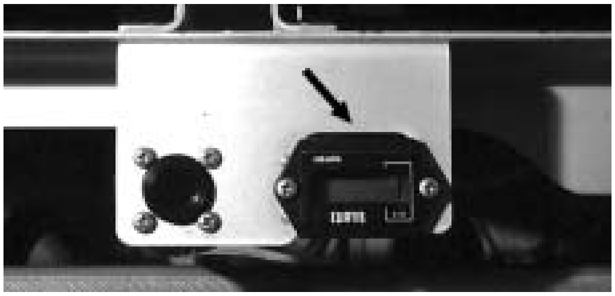


Figure 11-19 Hour Meter

The drive motor requires approximately 115 amps electrical power to drive the compressor. Therefore, if a generator trips off line in flight, the drive motor will shutdown and shed the high electrical load. This automatic load shedding provision, prevents the operating generator from exceeding its load limit and ensures adequate electrical power is available for essential flight equipment. The vapor cycle system may be operated on the ground with one engine generator operating or with a ground power unit (GPU) supplying electrical power.

APU BLEED AIR

If the APU is operating and the BLEED AIR switch is ON (BLEED VAL OPEN light illuminated), the APU will be supplying the majority of the bleed air requirements to the air conditioning system, even with the engines running at high power.

The APU BAV is installed in parallel with the engine flow control valves, but, supplies higher bleed air pressure until approximately FL200. The higher pressure through the BAV blocks engine bleed air through the engine flow control valves until higher altitudes are reached at which time APU bleed air becomes less efficient.

NOTE

Anytime the APU is running, service air pressure is always available whether the BAV is OPEN or CLOSED.

NORMAL OPERATION

PREFLIGHT

The Citation Excel air conditioning system is designed to eliminate excessive workload by the crew. The crew normally presets a minimum number of controls and the system operates automatically thereafter.

During preflight checks, the PRESS SOURCE selector is set to NORMAL and temperature control is set as desired. Normally, the CKPT TEMP SEL and the CABIN TEMP SEL selectors are set to the 12 o'clock position (75°F), and the center selector is set to either CKPT or CAB.

The vapor cycle air conditioner panel is normally set OFF. If supplemental air conditioning is desired, activate it after a ground power unit (GPU) is connected and supplying power or an engine is running and the generator is online. It may be operated in either AC/ON HI or LO, or WEMAC BOOST HI or LO.

The PRESS SOURCE selector may be left in NORMAL and the temperature selectors may remain as previously discussed during engine starts. However, the vapor cycle air conditioner should be turned off during engine starts.

After the engine(s) are started, bleed air commences flowing from the engines through the precoolers, ECU, and the cabin and cockpit air distribution system.

IN FLIGHT

In flight, the PRESS SOURCE selector is normally left in NORM, and the cabin and cockpit temperature selectors in AUTO, and temp set as desired. The cabin and cockpit zone temperatures are checked periodically to ensure they are in the comfort zone, if not, adjustment of the selectors is required.

In flight, the vapor cycle air conditioner may be operated up to 18,000 feet altitude. If the system is left on, it will automatically shut-

down climbing through 18,000 feet (AC/ON "green" indicator light extinguished).

NOTE

Use of MANUAL temperature control should be restricted to below FL310 in order to prevent possible overheating of the ACM, which would result in automatic acuation of emergency pressurization (**ACM O'HEAT and EMER PRESS** annunciators illuminating). This could occur if requesting cold temperatures, resulting in excess bleed air flow through the ACM.

GROUND OPERATION, HOT DAY

If warm environmental conditions exist, the vapor cycle air conditioner may be turned on and cold air should begin flowing through the forward evaporator/blower grille. Enhanced air cooling and flow should also be noticed through the overhead Wemac outlets, especially if HI is selected on the A/C control panel. If environmental conditions are not too hot and supplemental cold air is not required from the vapor cycle system, then the A/C selector may be switched to WEMAC BOOST only.

If increased air circulation is required in the cockpit, turn on the CKPT RECIRC switch to HI or LO located on the copilot's lower switch panel to blow air through the Wemac outlets on either side of the instrument panel.

The CKPT TEMP SEL and CABIN TEMP SEL selectors may be positioned to set cooler temperatures as desired.

NOTE

If the airplane is anticipated to remain parked during daylight in hot weather for an extended period, it is recommended that the cabin window shades be closed to reduce solar heat transfer. An optional exterior windshield cover may be used to reduce heat in the cockpit.

The cabin may be cooled prior to passengers boarding, by use of vapor cycle air conditioning operating from a generator (ensure generator amp load is not exceeded on the ground) or GPU power.

Airplanes with an optional APU installed (vapor cycle unit removed), may operate the APU to supply bleed air through the ECU for interior ground cooling.

GROUND OPERATIONS, COLD DAY

Heating the interior before passengers board, is best accomplished by starting the right engine with the PRESS SOURCE selector in NORM. Switching the CKPT and CABIN TEMP SEL knobs to MANUAL and selecting MANUAL HOT for 10 seconds, ensures the temperature control valves (TCVs) are full open (hot) for the cockpit and cabin. Turning on the CKPT RECIRC fan HI will increase air circulation in the cockpit. Operating the right engine above idle RPM increases temperature and airflow.

It is recommended that the cockpit be warmed to at least 50ºF as indicated on the cockpit temperature indicator before flight. This can be accomplished by taxiing the aircraft to a suitable area and increasing power above idle (approximately 60% N_2) to obtain duct supply temperatures of approximately 200ºF.

NOTE

The optional APU, if installed, may be used to generate a large amount of bleed air through the EPU to rapidily heat the interior.

ABNORMAL OPERATION

SMOKE OR ODOR

The PRESS SOURCE selector switch can be used to help isolate a source of smoke or odor that may be emanating from the air conditioning system. If the cockpit recirculating fan is on HI, it should be placed in LO or OFF, and the vapor cycle air conditioner or Wemac boost switch placed OFF. After a short period of time, If smoke or odor is still present, the PRESS SOURCE selector should be selected to LH and RH for approximately one minute each to determine if the smoke or odor dissipates. If smoke or odor is present in either LH or RH position, then the source of the smoke or odor is from that engine. If smoke or odor is still emerging with the PRESS SOURCE selector in LH and RH positions, the source is most likely from the ECU/ACM. In this case, positioning the PRESS SOURCE selector to EMER should eliminate the smoke or odor. Temperature will now be influenced by the left engine throttle.

NOTE

To determine if smoke or odor is emanating from an optional APU that is running with the BAV open, place the PRESS SOURCE selector OFF. After a minute, if smoke or odor is still present, shut down the APU.

ENGINE BLEED AIR OVERHEAT

If an engine bleed air overheat occurs, **BLD AIR O'HEAT** L or **R** annunciator illuminates "flashing", the PRESS SOURCE selector should be selected to the opposite engine and power reduced on the affected engine. The overheat condition was most likely caused by a defective precooler.

NOTE

If WING ANTI-ICE is ON, the respective wing anti-ice bleed air will automatically shut off if the **BLD AIR O'HEAT L** or **R** annunciator illuminates.

SUPPLY DUCT OVERHEAT

An air supply duct overheat will be identified by an **AIR DUCT O'HEAT CKPT** or **CAB** annunciator illuminating "flashing." The abnormal situation can normally be corrected by placing the affected system temperature selector (CKPT TEMP SEL or CABIN TEMP SEL) to MANUAL COLD and hold in this position (30 seconds maximum) until the annunciator extinguishes. If this procedure corrects the problem, the temperature selector should be returned to AUTO and a colder temperature selected. If the annunciator re-illuminates again in AUTO, select MANUAL mode and operate the affected system in MANUAL.

NOTE

Operating above 31,000 feet in MANUAL full cold may result in an ACM over temperature condition and shutdown (**EMER PRESS** and **ACM O'HEAT** annunciators illuminating).

If after operating in MANUAL full cold for a short period, and the respective **AIR DUCT O'HEAT CKPT** or **CAB** annunciator remains illuminated, place the PRESS SOURCE selector to LH or RH and reduce power on the corresponding engine as necessary to extinguish the light and control temperature (reduces total airflow).

TEMPERATURE CONTROLLER INOPERATIVE

If cabin or cockpit temperature cannot be controlled in AUTO mode of operation, the affected system temperature selector (CKPT TEMP SEL or CABIN TEMP SEL) should be selected to MANUAL. To ensure the affected TCV valve is not fully closed (full cold), hold the selector to full cold for a least 10 seconds and the actuate toward MANUAL HOT for at least 3 seconds.

NOTE

Operation in MANUAL, full cold, above 31,000 feet, particularly at low (climb) airspeed, may result in an ACM overtemperature and shutdown (**EMER PRESS** and **ACM O'HEAT** annunciators illuminating).

ACM OVERHEAT

If excessive pressure of the bleed air supply exists in the ACM or overheating of the ACM occurs, the ACM will automatically shutdown and emergency pressurization will automatically occur (**ACM O'HEAT** and **EMER PRESS** annunciators illuminate "flashing"). This abnormality can normally be corrected by adjusting the affected system (CKPT or CABIN TEMP SEL) selector in AUTO to a warmer setting or it may require operating in MANUAL. If selecting a warmer temperature in AUTO or MANUAL doesn't correct the situation, ensure the PRESS SOURCE selector is in LH, RH or NORM. If the annunciators remain on, then the PRESS SOURCE selector should be selected to EMER and cabin temperature will have to be controlled by the left throttle.

NOTE

Emergency pressurization uses very warm precooled bleed air from the left engine at high power settings.

ACM SHUTDOWN OR FAILURE

If an ACM failure or shutdown occurs, the air conditioning system will revert to emergency operation (**EMER PRESS** annunciator illuminates "flashing"). This will occur if the 5-amp NORM PRESS circuit breaker opens and the 5-amp EMER PRESS circuit breaker remains in (both CBs are located on the LH circuit breaker panel). If the NORM PRESS circuit breaker is in, adjusting both CKPT and CABIN TEMP SEL selectors to a warmer setting (may require MANUAL mode) should correct the problem. Increasing the temperature settings, deflects a portion of the bleed air from the ACM and directs it to the TCV valves, consequently, reducing bleed air pressure through the ACM.

If the **EMER PRESS** annunciator remains on, the PRESS SOURCE selector should be selected to EMER and then to RH, LH or NORM to attempt a manual reset. If the problem still exists after attempting a reset, it will require selecting EMER with the PRESS SOURCE selector and control temperature with the LH throttle. Opening the overhead Wemacs, turning Wemac boost on HI, and selecting the CKPT RECIRC fan switch to HI, will enhance air circulation.

NOTE

Emergency pressurization uses very warm precooled bleed air from the left engine at high power settings.

QUESTIONS

1. in the event an **ACM O'HEAT** annunciator illuminates, what if any other indications will occur while airborne?
   A. **AIR DUCT O'HEAT** illuminates.
   B. **EMER PRESS** illuminates.
   C. **BLD AIR O'HEAT** illuminates.
   D. Engine fire light illuminates.

2. The 35°F low temp sensor associated with the water separator remains operational:
   A. If EMER is selected with the PRESS SOURCE selector.
   B. If MANUAL mode temperature control is selected.
   C. If main DC power is lost.
   D. If the A**CM O'HEAT** annunciator illuminates.

3. If the **AIR DUCT O'HEAT CKPT** annunciator illuminates:
   A. The ECU normal flow control valves will close.
   B. Air in the overhead air duct is too warm.
   C. Air flow into the cockpit under floor duct is too hot.
   D. Air from the ECU is too warm.

4. Air entering the cabin through the overhead ducts is:
   A. Temperature regulated.
   B. Recirculated cabin air.
   C. Fresh ram air.
   D. Unregulated ACM cold air.

5. Temperature Control Valves (TCV) provide:
   A. Mixing of ram air with cold air from the ECU.
   B. Increased or decreased air flow into the cabin and cockpit air distribution ducts.
   C. Automatic temperature control only.
   D. Hot bleed air from the precoolers to mix with cold air from the ECU.

6. The temperature control panel provides:
   A. Separate automatic and manual temperature control for the cabin and the cockpit.
   B. Automatic temperature control only.
   C. Manual temperature control only.
   D. Provides temperature indications in a digital format in degrees Celsius.

CHAPTER 12
PRESSURIZATION

CONTENTS

ILLUSTRATIONS

CHAPTER 12 PRESSURIZATION

INTRODUCTION

The EXCEL pressurization system is used to maintain a lower cabin altitude than actual airplane altitude. This is normally accomplished by ducting conditioned engine bleed air from the ECU into the cabin and controlling the amount of air escaping. The EXCEL pressurization and air conditioning systems normally employ common airflow from the ECU.

GENERAL

Cabin pressure is controlled pneumatically by a digital microprocessor. Cabin air inflow from the ECU varies from 20lbs/per/minute minimum at sea level to a minimum of 12 lbs/per/minute at 45,000 feet. The system has the capability of pressurizing the cabin to a maximum differential pressure of 9.5 psid however, differential pressure is normally maintained at 9.3 psid by the controller. Differential pressure of 9.3 psid equates to a cabin altitude of sea level at an airplane altitude of approximately 24,000 feet, and 6,800 feet at an altitude of 45,000 feet.

The Cabin Pressure Control System consists of a Digital Cabin Pressure Controller, primary and secondary outflow valves, a MANUAL TOGGLE VALVE, an AUTO/MANUAL switch and an EMER DUMP switch. The digital controller commands the outflow valves

to open or close as required to control cabin pressure and cabin altitude rate of change. The controller, AUTO/MANUAL switch, MANUAL TOGGLE VALVE and EMER DUMP switch are located on the cockpit tilt panel (Figure 12-1). A direct reading Cabin Altitude/Cabin Differential Pressure gauge is mounted adjacent to the digital control head. The primary and secondary outflow valves are located on the lower base of the aft pressure bulkhead to allow cabin air to exhaust into the tail cone.

OUTFLOW VALVES

Each outflow valve has a pneumatic control chamber on the front side (cabin side) of a diaphragm. The diaphragms cover 4-inch diameter grills. In the unpressurized state, the diaphragms are halfway open (neutral). The outflow valves are opened or closed by increasing or lowering control chamber pressure to position the diaphragms (Figure 12-2). As control chamber pressure is increased or decreased, the outflow valves are positioned toward open and closed due to increasing/ decreasing differential pressure acting on the aft side (tail cone side) of the diaphragm.

The primary outflow valve is equipped with two solenoids (climb and dive solenoids) controlled by the digital controller, a vacuum ejector, a Maximum Delta-P (DP) Limiter, and a Maximum Altitude Limiter. The limiters provide backup redundancy to protect crew and passengers in the event control malfunctions occur.

The secondary outflow valve is slaved to the primary outflow valve by a connecting tube to each control chamber to move both outflow valves simultaneously. The secondary outflow valve is similar to the primary outflow valve but does not contain climb or dive solenoids, or a vacuum ejector, but does have a maximum DP limiter

Figure 12-1 Pressurization Control Panel

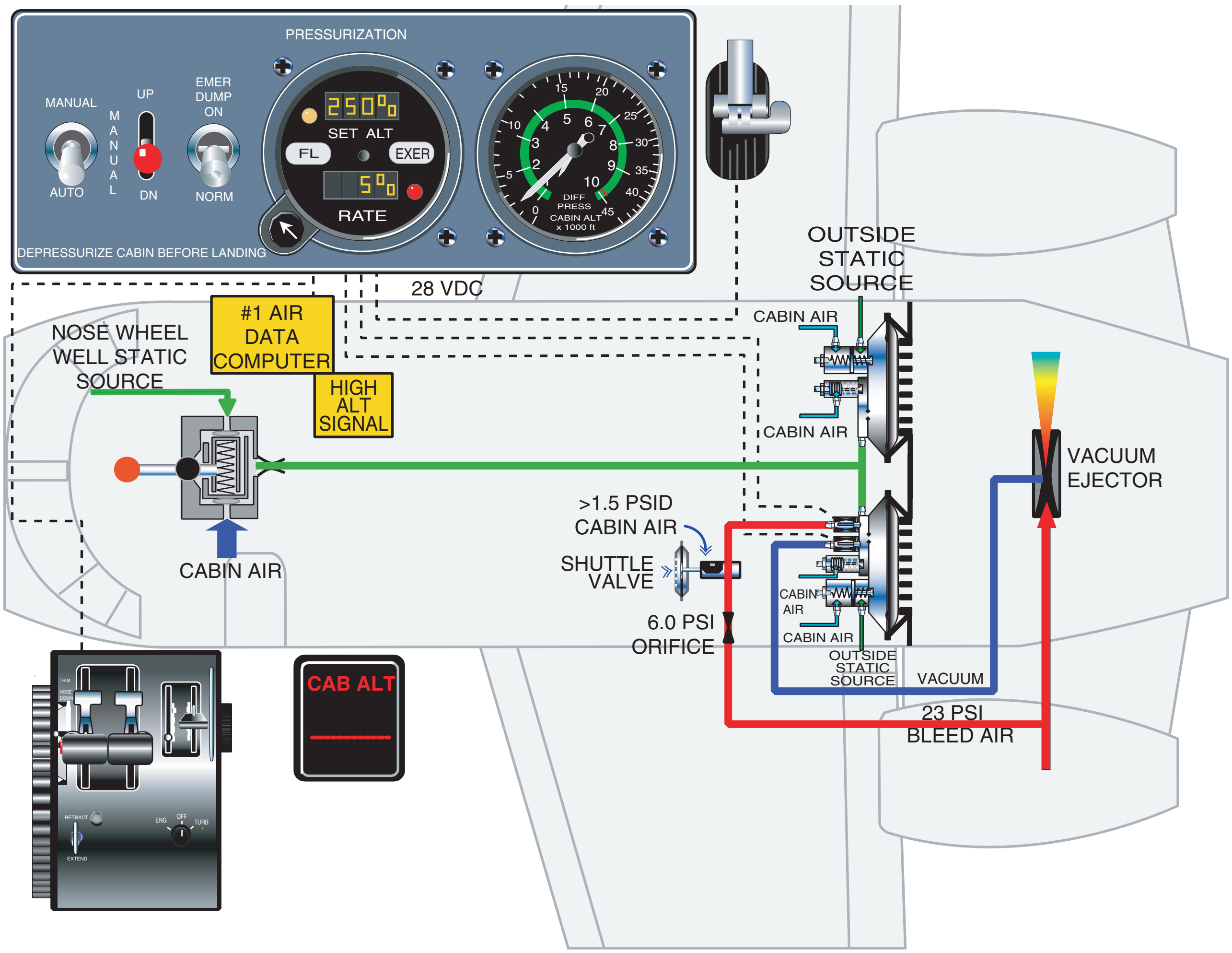


Figure 12-2 Pressurization System

and a maximum cabin altitude limiter to protect the cabin in the event of a primary outflow valve failure.

CLIMB/DIVE SOLENOIDS

The primary outflow valve has two normally closed solenoids (one dive solenoid and one climb solenoid) to control air entering or evacuating the control chamber. The climb solenoid when energized "open" allows air pressure in the control chambers to exit to the service air vacuum ejector built into the primary outflow valve (Figure 12-2). This action causes differential pressure to increase and force both outflow valves toward the open position.

The dive solenoid when energized "open" allows service air regulated to 6.0 psi to enter the control chamber to drive both outflow valves toward the closed position. A shuttle valve will allow filtered cabin air into the supply line and shutoff 6.0 psi service air to the control chamber when cabin DP exceeds 6.0 psi.

The solenoids are controlled by the digital controller which sends short bursts of main DC electrical power to cause the solenoids to momentarily open and generate gradual pressure changes to control the outflow valves. The system is designed to respond rapidly to small cabin pressure variations and correct them before passengers and crew detect any discomfort. The solenoids are never open simultaneously.

CABIN LIMITERS

Both outflow valves have Maximum DP Limiters and Maximum Altitude Limiters attached which prevent the cabin from exceeding maximum differential limits and cabin altitude limits.

The Maximum DP Limiters connect to the cabin side of the outflow valve control chambers. The control chamber diaphragm has cabin pressure exerted on one side and tailcone pressure on the other side. When differential pressure on the cabin side exceeds 9.5 ± 0.1 psid, the Max DP Limiter will open and let the outflow valve control chamber vent to outside ambient through static ports in the tail cone. This will cause the outflow valve to move toward the open position and reduce cabin pressure. The Max DP Limiters can override a dive solenoid signal, a MANUAL TOGGLE VALVE signal, a Maximum Altitude Limiter signal, or ruptured/leaking plumbing connections.

Maximum Altitude Limiters are plumbed to the outflow valve control chambers and incorporate sealed bellows that expand with decreasing cabin pressure (cabin altitude increasing). At a preset absolute pressure, the bellows expand to a point that unseats a poppet valve and allows cabin air into the outflow valves control chambers. This increasing air pressure in the control chamber causes the outflow valve to move toward the closed position and increase cabin pressure. The set point of the Max Altitude Limiters is 14,500 ± 500 feet. The Max Altitude Limiters are such that either outflow valve can override a climb or dive solenoid signal, and override the MANUAL TOGGLE VALVE signal but not the Maximum DP Limiters.

AUTOMATIC MODES (AUTO SCHEDULE)

The primary operating method is the Auto Schedule Mode. The auto schedule mode is determined with the AUTO/MANUAL switch in AUTO (Figure 12-1 and 12-2). Automatic mode of cabin pressurization scheduling is a function of aircraft altitude, cabin altitude, throttle position and aircraft on the ground or in flight.

NOTE

Cabin differential pressure is normally regulated between 0 and 9.3 psid while operating in auto schedule.

The controller receives aircraft altitude from the pilot's micro air data computer (No. 1 MADC), and cabin altitude is sensed internally. Throttle position and ground/flight status are transmitted to the controller by throttle position switches and a squat switch on the left landing gear.

The pressurization controller is comprised of two digital windows marked SET ALT and RATE, FL and EXER buttons, and a SET ALT rotary knob (Figure 12-1).

With the AUTO/MANUAL switch (Figure 12-1) in AUTO, the pilot selects destination field elevation (Set Landing Altitude – SLA). SLA for destination is normally accomplished during the before taxi checklist, however, it may be set at any time prior to landing.

The controller logic maintains the lowest cabin altitude practical for the aircraft throughout the flight. Each phase of flight determines the modes of operation the controller follows. The controller will be in one of five modes of operation during a typical flight. The auto schedule modes of operation are: ground mode, takeoff mode, flight mode, descent mode or high altitude mode. The only crew action required is to set the SLA, normally prior to take-off, and the modes of flight are automatic throughout the flight.

In flight, the controller continually generates an auto schedule based on departure field elevation, maximum aircraft altitude reached in the current flight, and the operator input destination field elevation (SLA). The controller determines the pressure rate of change and the cabin altitude based on the programmed Auto Schedule Boundary (Figure 12-3). The

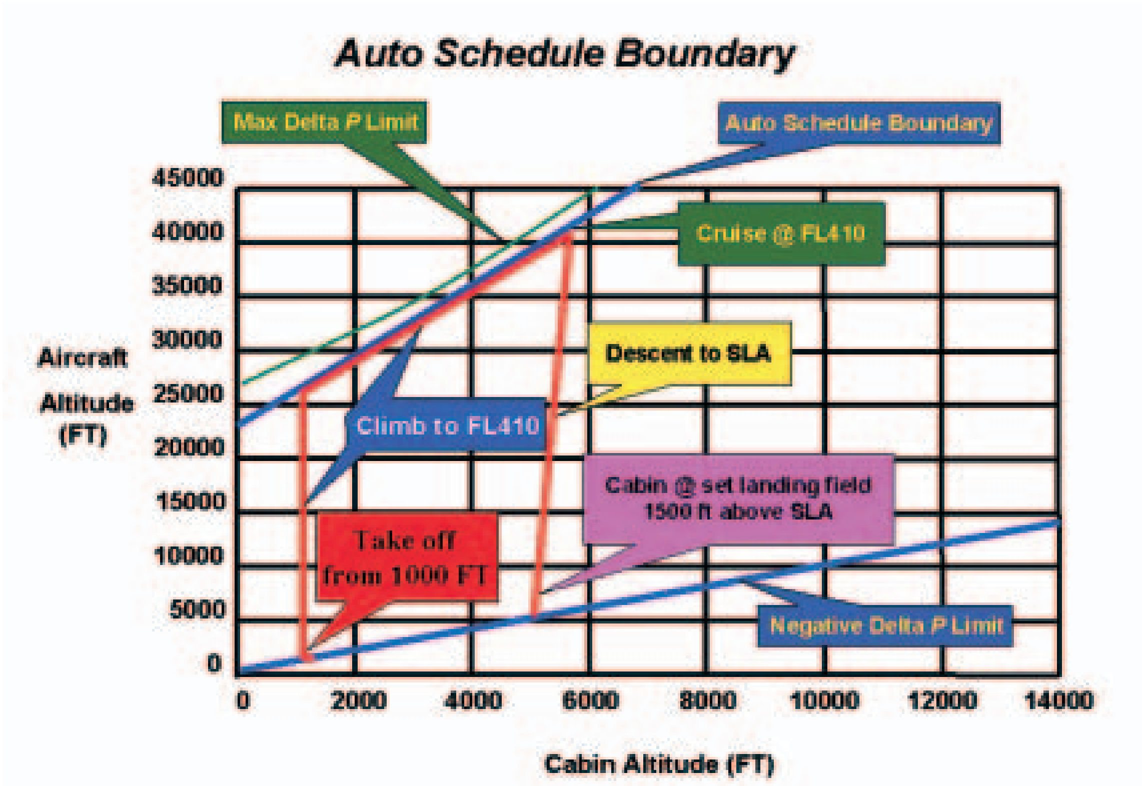


Figure 12-3 Auto Schedule Boundary

cabin climb and dive rates are limited to +600/-500 fpm respectively while operating in the auto schedule mode. The controller continually sends DC pulses to the primary outflow valve dive and climb solenoids to obtain specific cabin pressure.

GROUND MODE

With the aircraft on the ground (left squat switch), throttles below approximately 85% Throttle Position Angle (TLA), and the main DC electrical system powered, the controller operates in the ground mode. In this mode, the controller signals the climb solenoid open continuously to hold both outflow valves open. This will minimize positive cabin pressure on the ground.

TAKEOFF MODE

The controller enters the takeoff mode as the throttles are advanced to the takeoff power setting (above approximately 85% TLA), and the left squat switch is in the ground-on-ground position. The controller sends short bursts of electrical power to the dive and climb solenoids to gradually drive the outflow valves from full open toward the closed position at a rate of approximately –100 fpm. During a normal takeoff ground run the cabin will normally descend to between 50 and 100 feet below takeoff field elevation. From this point on, the controller is actively controlling cabin pressure throughout the flight.

FLIGHT MODE

The flight mode is entered immediately as the aircraft becomes airborne (left squat switch weight-off-wheels). The controller receives aircraft altitude from the pilot's micro air data computer (No. 1 MADC), cabin pressure from an internal sensor, and the Set Landing Altitude (SLA) in the SET ALT window and controls

cabin pressure as determined from a programmed auto schedule input (Figure 12-3). Operation of the auto schedule mode will be discussed in detail during the discussion of a typical flight profile below.

DESCENT MODE

The descent mode is entered as the aircraft descends 500 feet below its cruise altitude. The cabin descent rate is determined by the programmed auto schedule and takes into account the aircraft descent rate and the SLA. Cabin altitude will be at the SLA as the aircraft descends to 1,500 feet above the landing field elevation and remain at SLA as the aircraft continues its descent for landing. Upon landing, the controller returns to the ground mode (outflow valves held full open) to ensure the cabin remains depressurized during taxi and shutdown.

TYPICAL FLIGHT PROFILES

A typical flight profile is shown in Figure 12-3.

GROUND MODE

The example shown in Figure 12-3 starts with the aircraft on the ground at 1000 feet airfield elevation. With the airplane on the ground, left squat switch ground-on-ground, throttles below approximately 85% N_2, and main DC electrical power, the controller signals the climb solenoid open holding both outflow valves fully open (ground mode) to minimize cabin pressure. Cabin altitude should be the same as airfield altitude.

TAKE-OFF MODE

During take-off, as the throttles are advanced to take-off power, the controller enters the takeoff mode and cabin pressure begins to slowly increase. The controller sends short bursts of electrical power to the climb and dive solenoids to cause the outflow valves to gradually drive toward the closed position. The system is programmed to drive cabin pressure down at a rate of 100 fpm for two minutes. However, the takeoff roll should normally require much less time. During a normal takeoff roll, the cabin pressure should only increase to approximately 50 to 100 feet below field elevation. The takeoff mode prevents cabin pressure “bumps” during the takeoff roll and during the transition to flight by controlling cabin pressure throughout the takeoff.

FLIGHT MODE

As the aircraft leaves the ground, left squat switch weight-off-wheels, the controller switches to the flight mode. During the climb phase, the cabin remains at the point it achieved during the takeoff roll, approximately 50 to 100 feet below takeoff field elevation as shown in Figure 12-3, until cabin differential pressure reaches the auto schedule boundary as programmed into the controller (rate window indicates zero cabin climb rate). This occurs when the aircraft reaches approximately 26,000 feet as shown in the example depicted in Figure 12-3. As the aircraft climbs to a cruise altitude of FL 410, the cabin altitude climb rate will follow the auto schedule and reach an altitude of approximately 5,600 feet as the aircraft reaches FL 410 (Figure 12-3).

If the aircraft’s rate-of-climb from FL 260 to FL 410 averages 1,000 fpm, the cabin rate window would indicate approximately +300 (+300 fpm). When the cabin altitude reaches the autoschedule boundary of 9.3 psid, the differential pressure gauge needle should remain at this pressure differential during the remainder of the climb to cruise flight level and during the cruise portion of the flight.

NOTE

The controller is programmed to limit cabin altitude climb and descent rates to +600/-500 fpm respectively.

NOTE

If the aircraft continued a climb to its maximum altitude limit of FL 450, the cabin altitude would follow the autoschedule and climb to a maximum of 6,770 feet.

In the example (Figure 12-3), the cabin remains at 5,600 feet during the cruise portion at FL 410.

DESCENT MODE

As the aircraft descends 500 feet below cruise flight level, the controller enters the descent mode and begins to rate the cabin down toward the Set Landing Altitude (SLA), 5000 feet in the example (Figure 12-3). If the aircraft descends at approximately 2,000 fpm, the cabin altitude will descend approximately 30 fpm. The rate window should indicate -0^0$_0$.

NOTE

If an extremely high descent rate is conducted, example, an emergency descent, the controller will recognize the high rate and continually compensate (update) the system to ensure the aircraft will not fly through the cabin altitude and cause an immediate depressurization.

When the aircraft descends to approximately 1,500 feet above the SLA (in the example, 6,500 feet), the cabin altitude will be at SLA (5,000 feet). The cabin will remain at the SLA altitude until landing. On landing, as the left squat switch activates, ground-on-ground, the controller will switch to the ground mode to ensure the cabin is depressurized.

HIGH ALTITUDE MODE – AUTO SCHEDULE

The high altitude mode is somewhat of a misnomer. There is not a high altitude switch, button, etc. Selecting a high altitude airport destination, above 8,000 feet in the SLA window prior to departure, alerts the system to operate in the high altitude landing mode after takeoff. Figure 12-4 is a graphic example of a high altitude landing (14,000 feet) from a low altitude departure (3,000 feet).

After takeoff, the controller initially operates in the high altitude landing mode, which causes the cabin altitude to increase at 600 fpm until the cabin reaches approximately 8,000 feet. The cabin remains at 8,000 feet during the remainder of the climb, enroute cruise and descent through 24,500 feet.

NOTE

If a long extended flight is planned from a low altitude airport to a high altitude airport above 8,000 feet, consideration should be given to selecting the departure elevation in the SLA window. This will allow the system to operate in a normal auto schedule mode and maintain a lower cabin altitude during climb and enroute. However, the destination airport should be set in the SLA window approximately 30-45 minutes prior to descent from cruise altitude. The controller is programmed to ascend the cabin to 8,000 feet at 100 fpm.

High Altitude Landings

Descending through 24,500 feet the cabin climbs as required to arrive at destination field elevation as set in the SLA window at 1,500 feet above SLA (15,500 feet, Figure 12-4). Depending on the aircraft descent rate and airport field elevation (set in SLA window), the cabin will climb as high as 2,500 fpm if required to reach SLA 1,500 feet prior to landing (see Figure 12-5).

NOTE

Operating below 24,500 feet. approaching a high altitude airport for landing, the **CAB ALT** annunciator will not illuminate unless cabin altitude exceeds 14,500 feet.

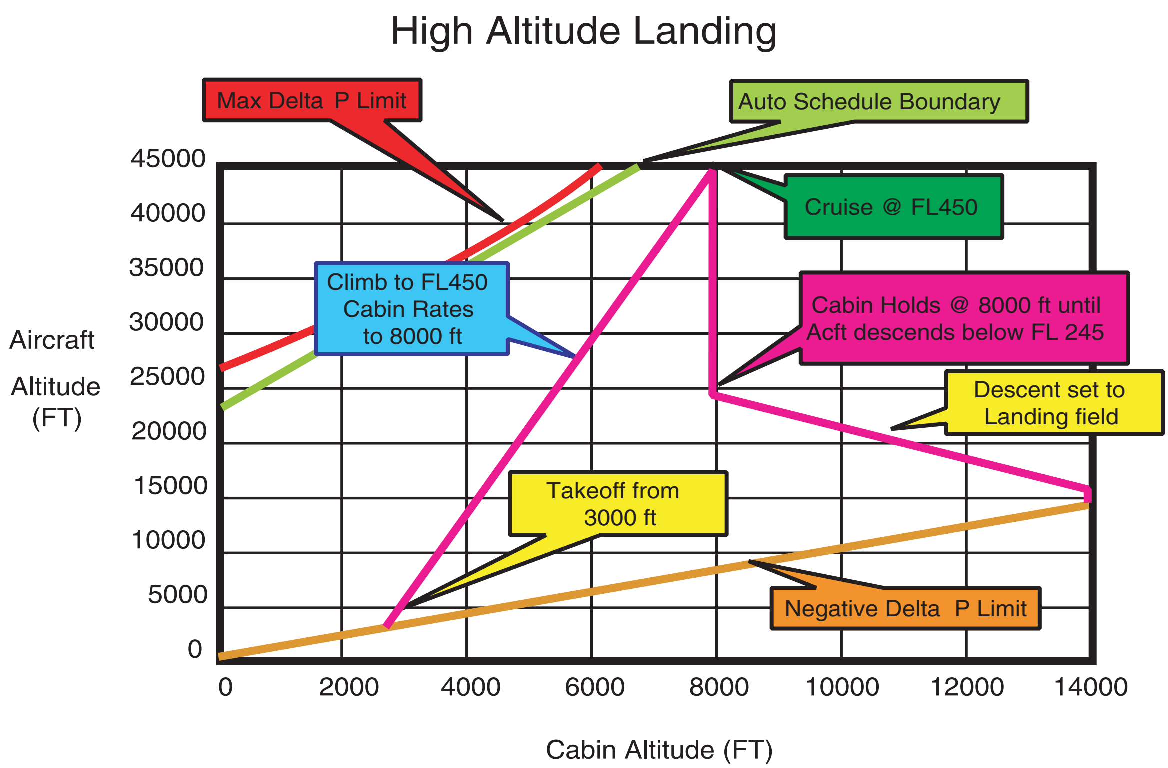


Figure 12-4 High Altitude Landing

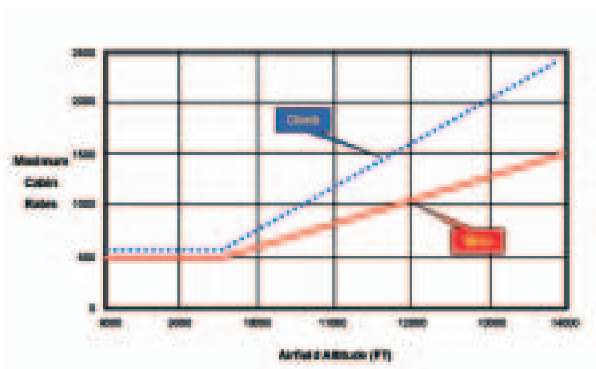


Figure 12-5 Modified Cabin Rate Limits High Altitude Mode.

During the final approach to landing, cabin altitude will remain at SLA and depressurize just prior to landing.

If the controller malfunctions or the SLA window is set below destination elevation and the cabin is pressurized upon landing, the cabin will begin depressurizing at a rate of 1,000 fpm for 30 seconds. If still pressurized after 30 seconds, the system will return to the ground mode, throttles below approximately 85% N_2, and any residue cabin pressure will be dumped.

High Altitude Departures

An example of a high altitude departure is shown in Figure 12-6.

The flight begins with the aircraft departing from an airfield at 14,000 feet elevation and landing at a 3,000 feet elevation airport. The destination field elevation is set in the SLA window prior to departure.

After takeoff and climb to an enroute altitude, the cabin altitude begins a dive rate up to 1,500 fpm if required (Figure 12-5), to reach 8,000 feet cabin altitude by 25,000 feet aircraft altitude.

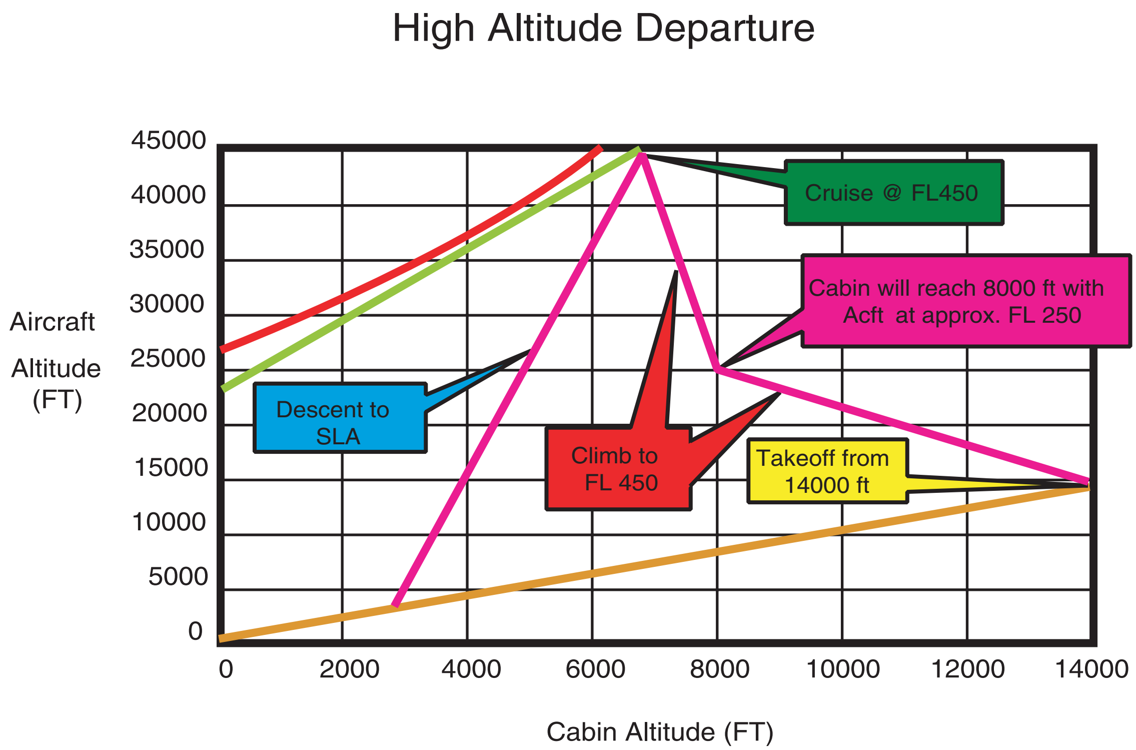


Figure 12-6 High Altitude Departure

NOTE

Operating below 25,000 feet departing a high altitude airport, the **CAB ALT** annunciator will not illuminate unless cabin altitude exceeds 14,500 feet.

After climbing through 25,000 feet, the cabin altitude will continue to decrease at 100 fpm towards the autoschedule boundary or SLA which ever is higher. In Figure 12-6, the autoschedule boundary is intercepted at FL 450 (cabin altitude 6,770 feet). The cabin altitude will hold at 6,770 feet during cruise at FL 450.

NOTE

If the aircraft experiences an emergency that requires an emergency return to the departing airport and levels at an altitude below 25,000 feet, after one minute, the cabin altitude will be rate controlled down at 100 fpm to the altitude set in the SLA window. To prevent landing pressurized, reset the departure altitude in the SLA window or set prior to departure.

During descent, as the aircraft descends 500 feet below cruise altitude, the cabin begins to rate down towards SLA (3,000 feet). The cabin should reach SLA when the aircraft is 1,500 feet above landing field elevation and maintain this altitude until landing.

ISOBARIC MODE

The isobaric mode cannot be entered directly by the pilot in flight. The isobaric mode is a standby mode if the signal from the number 1 air data computer is lost. If the signal to the No. 1 MADC is interrupted or lost, the controller will automatically switch from Auto

Schedule to Isobaric Mode. A yellow warning indicator will illuminate on the face of the controller to advise the crew to switch to Isobaric Control. The SLA window will revert from Selected Landing Altitude to Flight Level (current aircraft altitude) (Figure 12-7).

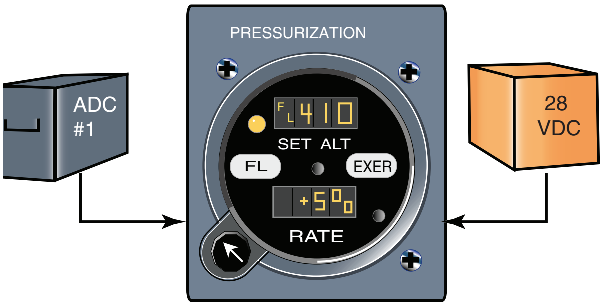


Figure 12-7 Isobaric Mode — FL

If the aircraft is climbing or descending during a change to isobaric mode, the pilot should manually select the desired level off flight level in the SET ALT window with the set knob. If the system reverts to isobaric mode during level flight and a clearance is issued to change flight level, set the new flight level in the SLA window prior to transition to a climb or descent.

The controller uses the selected Flight Level to control cabin pressure rate of change and cabin pressure altitude to maintain near the auto schedule boundary.

Prior to or during descent to landing, the crew should press the "FL" button on the controller face, which will replace Flight Level (FL) with Cabin Altitude (CA) in the SET ALT window. The current cabin altitude will be displayed (Figure 12-8). Using the set knob,

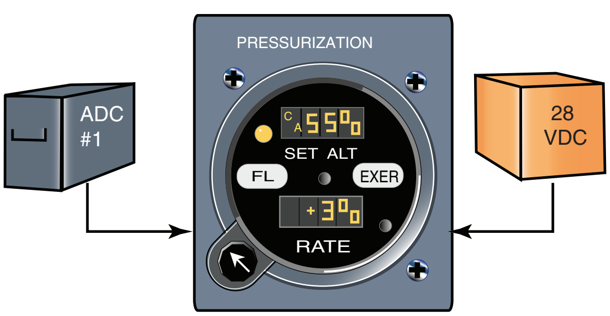


Figure 12-8 Isobaric Mode — CA

set the desired landing field elevation in the window (CA). The controller then controls the cabin pressure rate of change to rate the cabin toward the selected CA in the window.

NOTE

The maximum cabin rate of change is limited to +600/-500 fpm while in the Isobaric Control Mode regardless of the departure or landing airport elevations (no high altitude mode).

The crew may flip-flop Flight Level and Cabin Altitude displays by pressing the FL push button on the controller. If the No. 1 MADC resumes operation, the controller will automatically switch back to AUTO SCHEDULE CONTROL MODE and the yellow light will extinguish.

MANUAL MODE

Manual Mode is entered if the controller fails or MANUAL is selected with the AUTO/MANUAL switch on the pressurization control panel (Figure 12-9). A controller failure is annunciated with a "red" indicator (same location as the "yellow" isobaric indicator) on the face of the controller and the RATE and SET ALT windows go blank, and the outflow valves will remain in their last commanded position. If this failure occurs in flight, cabin pressure may be controlled using the MANUAL TOGGLE, "Cherry Picker." Placing the AUTO/MANUAL switch to MANUAL removes electrical power from the controller and simultaneously removes power from the climb and dive solenoids.

The MANUAL TOGGLE (Cherry Picker) UP/DN lever (Figure 12-9) adjacent to the AUTO/MANUAL switch is used to directly control pneumatic pressure to the outflow valves. Holding the "Cherry Picker" UP vents the outflow valves control chambers to outside atmosphere with a plumbed connection from a "T" fitting between the two outflow valves

Figure 12-9 Emergency Dump Switch

to a static port located in the nose gear actuator tunnel. Venting the outflow valves to outside pressure (lower than cabin pressure) in the nose gear wheel well has the same effect as supplying vacuum to the control chambers, thus causing the outflow valves to move toward the "open" position (Figure 12-2).

Holding the "Cherry Picker" DN allows cabin air pressure to enter the control chambers and move the outflow valves toward the "closed" position (Figure 12-2).

NOTE

The MAX DP Limiters will override the MANUAL TOGGLE UP/DN valve and prevent cabin pressure from exceeding 9.5 ± 0.1 psid.

The MANUAL TOGGLE valve is orificed so that it cannot overpower the Max DP Limiters/Cabin Altitude Limiters. However, it can override the climb and dive solenoids. The amount of cabin altitude change and cabin rate of change is controlled by the time the "Cherry Picker" is held in the UP or DN position. The longer the Cherry Picker is held DN, the faster the rate of descent increases, alternately, the longer the Cherry Picker is held UP, the faster the rate of climb increases. Since the toggle valve uses cabin pressure to close the outflow valves, it is less effective at low cabin pressure than at higher cabin pressure.

NOTE

To check MANUAL MODE on the ground prior to dispatch, cabin pressure "zero" psid, advance power above 60% N_2 and hold the "Cherry Picker" DN for a few minutes. After a few minutes, cabin pressure should begin to slowly increase. This procedure will be necessary in order to dispatch with the system in isobaric mode to ensure a backup means of controlling pressurization (refer to Excel MMEL).

EMERGENCY DUMP

Refer to the Emergency Dump Switch (Figure 12-9).

The EMER DUMP ON/NORM switch is used to rapidly dump cabin pressure if required. The switch requires main DC electrical power and ship's vacuum to dump cabin pressure. Activating the switch ON powers the climb solenoid OPEN and vents the outflow valve control chambers to ships vacuum (Figure 12-2). If the airplane altitude is above 14,500 feet when the emergency dump switch is activated, cabin altitude increases rapidly to approximately 14,500 feet at which time the Max Altitude Limiters will not let cabin altitude exceed limits, 14,500 ± 500 feet.

As the cabin rapidly approaches 14,500 feet the emergency pressurization valve should open (EMER PRESS annunciator illuminates) and drive cabin altitude down until the emer valve closes. The emer valve may cycle open and closed as cabin pressure changes.

BUILT-IN TESTS

DIAGNOSTIC TESTS

The digital controller incorporates two sets of built in diagnostics tests. The first test (BIT) is an internal check of the controller and is performed continuously during operation. If an internal fault is detected by the BIT during operation, all power is removed from the controller and a "red" indicator on the face of the controller illuminates and the windows go blank.

The second is a diagnostic check (BITE) of the controller and can detect some faults outside the controller, normally referred as the maintenance test mode. The BITE test is provided to assist maintenance in troubleshooting faults. The BITE test is activated by maintenance personnel while the aircraft is on the ground by pressing a recessed push-button with a slender nonconductive tool located between the FL and EXER button.

EXERCISE TEST

The crew or maintenance personnel may check the validity of the control system by pressing and holding the EXER button on the control panel (Figure 12-9) for two minutes. During the test, the controller will command the dive and climb solenoids to activate and move the outflow valves toward the closed position and pressurize the cabin to 200 feet below the current field elevation at a rate of 100 fpm (rate window will indicate -100). The test will complete and stabilize the cabin at -200 feet below current field elevation. Releasing the button, the controller will conduct a display test and gradually reopen the outflow valves to full open and slowly depressurize the cabin. The display test will be indicated by brightly illuminated dot patterns in both the SLA and RATE windows.

Pressing the EXER button in flight only conducts a lamp test.

EMERGENCY/ ABNORMAL OPERATION

OVERPRESSURIZATION

If overpressurization occurs (DIFF PRESS/CABIN ALT gauge, small needle exceeds 9.5 psid), the pressurization control system should be selected to MANUAL, and the manual toggle lever utilized to control cabin altitude within differential pressure limits.

If MANUAL control does not correct the overpressurization situation, the PRESS SOURCE selector should be placed to LH or RH and the corresponding throttle used to control cabin pressure. If this procedure still does not control cabin pressure, the crew and passengers need to go on oxygen and the PRESS SOURCE selector, selected to OFF, and begin an immediate descent. The cabin should begin climbing at approximately 500 - 600 fpm.

If the cabin is still overpressurized, the EMER DUMP switch will have to be activated.

NOTE

EMER DUMP requires main DC electrical and ship's vacuum. The cabin climb climb rate will normally exceed 6,000 fpm.

As the cabin approaches 14,500 feet, the passenger oxygen masks will automatically drop from the overhead passenger compartments, if they were not previously dropped manually (refer to Chapter 17, OXYGEN SYSTEM), and the ECU will automatically revert to emergency (EMER PRESS annunciator illuminates "flashing").

CABIN DECOMPRESSION (CAB ALT WARNING ANNUNCIATOR ON)

If cabin altitude begins an uncontrolled climb, the red CAB ALT warning annunciator will illuminate "flashing" at 10,000 feet (normal pressurization modes) or 14,500 feet (high altitude pressurization modes below FL245). The flight crew should immediately don oxygen masks at 100% oxygen, and the microphone switches on the lower right and left sides of the instrument panel should be selected to MIC OXY MASK. An emergency descent should be initiated as required, depending on aircraft altitude.

After the above procedures are initiated, the passengers must be checked to ensure they are receiving oxygen, masks dropped manually or automatically and the lanyard cords are pulled. The transponder should be selected to EMERGENCY.

NOTE

Consideration should be given to placing the PRESS SOURCE selector to EMER as soon as possible. If a duct failure occurred in the tailcone from the ACM to the aft pressure bulkhead, the cabin will experience a loss of air pressure and will begin a climb at approximately 500 - 600 fpm. Selecting EMER, will provide a direct source of bleed air from the left engine to the cabin underfloor network allowing the cabin to begin pressurizing.

NOTE

If the cabin climbs to 14,500 feet, emergency pressurization will activate automatically.

CABIN PRESSURIZATION CONTROLLER FAILURE

If main DC electrical power is available and the cabin pressurization controller fails, the a red indicator light will illuminate on the face of the controller and the SLA and RATE windows will go blank. The cabin should remain at its previous altitude prior to the failure.

If cabin altitude is not being maintained (DIFF PRESS/CABIN ALTITUDE gauge displays a climb), selecting MANUAL control and using the "Cherry Picker" should regain control of cabin pressure. Controlling cabin pressure with the "Cherry Picker," is a sensitive operation. It is not advisable to hold the manual toggle in either the UP or DN position too long. Positioning it to the UP or DN position momentarily and releasing, will allow a smoother cabin rate. If the climb or descent rate is too abrupt, momentarily tapping the toggle to the opposite position will slow the rate.

CAUTION

CABIN MUST BE MANUALLY DEPRESSURIZED PRIOR TO LANDING.

If cabin altitude is not arrested by 10,000 feet, (CAB ALT annunciator flashing), the crew should don oxygen masks and select microphone switches to MIC OXY MASK (extreme lower sides of the instrument panel).

If cabin altitude is not arrested by 14,000 feet, emergency descent should be initiated and the passenger oxygen masks dropped manually (PASS OXY valve, ON). The passengers should be checked visually to ensure they are receiving oxygen (masks on and lanyard cords pulled).

NOTE

Emer Press will automatically activate when cabin altitude exceeds approximately 14,500 feet, and will shut off when cabin altitude descends below approximately 12,000 feet.

If cabin pressure is maintained, but the amber fail indicator light on the pressure controller is illuminated, indicates probable loss of the number one air data computer and the auto schedule function is inoperative.

NOTE

If the number one MADC fails, all air data information on the pilot's PFD will display blank information with red Xs and/or dashes, i.e., airspeed, altimeter, and vertical speed displays.

In the "isobaric mode" of operation, the controller SLA window will display "FL" and current aircraft flight level. Prior to descending or climbing to a new flight level, or if the controller fails to the "isobaric mode" while climbing or descending to a specified flight level, selecting the desired flight level in the SLA window with the set knob will allow the controller to control cabin pressure rate of change and cabin pressure altitude to maintain near the auto schedule boundary.

Should a new cabin altitude be desired, pushing the FL button on the controller will change the "FL" display in the SLA window to "CA" (cabin altitude). Selecting a new cabin altitude reference in the SLA window will allow the controller to control cabin pressure rate of change near the auto schedule reference for climb and descents.

Prior to descent for landing, set the destination airport field elevation in the SLA window with "CA" displayed.

LIMITATIONS

Normal Cabin Pressurization Limitations
0.0 to 9.3 psi, +0.1 or -0.1 psi differential

Pressure Relief Valve
9.5 PSI, +0.1 or -0.1 psi differential

Pressure Gauge Red Line
9.7 PSI, +0.1 or -0.1 psi differential

QUESTIONS

1. Cabin pressure is normally maintained by:
   A. Controlling the amount of air entering the cabin.
   B. Controlling the amount of air escaping the cabin.
   C. Modulating the temperature of the Cabin Temperature Controller.
   D. Manipulating the throttles.

2. The normal outflow valve control modes are:
   A. Ground Taxi Mode.
   B. Takeoff Mode.
   C. Flight, Descent and Landing Modes.
   D. All of the above.

3. As both throttles are advanced above 85% N_2 on the ground, the outflow valves slowly close, driving cabin pressure down below field altitude. This is the:
   A. Flight Mode.
   B. Ground Mode.
   C. Takeoff Mode.
   D. None of the above.

4. What OPENS the two outflow valves to climb the cabin?
   A. AUTO uses the Digital Cabin Pressure Controller to meter ejector vacuum to the climb solenoid of the primary valve
   B. The Cabin Dump Switch sends DC power to the climb solenoid and the cabin rises to 14,500 feet (Cabin Altitude Limit Valve).
   C. With Manual Switch selection and using the manual toggle to manually meter ambient low pressure air to both outflow valves.
   D. All of the above.

5. What CLOSES the two outflow valves to dive the cabin?
   A. AUTO uses the Digital Cabin Pressure Controller to meter 6.0 psi Service Air Pressure or cabin air pressure to the dive solenoid.
   B. In the Manual Switch position, using the “Cherry Picker” to manually meter cabin pressurized air to the Secondary Valve only.
   C. Both A and B.
   D. None of the above.

6. If control vacuum becomes excessive in flight, cabin altitude:
   A. Explosively decompresses to cruise altitude
   B. Will remain at present altitude.
   C. Rises to the Maximum Altitude Limit Valves setting of 14,500 feet, and cabin pressure enters the outflow valves reducing vacuum effect and cabin stops climbing at approximately14,500 feet.
   D. Decreases to a value as determined by the Max Differential Pressure Relief Valve setting.

7. The source of bleed air for cabin pressurization when the EMERG PRESS light is illuminated in flight is:
   A. Vapor cycle air.
   B. The left engine.
   C. Either or both engines.
   D. Ram air flow.

8. The digital pressurization controller modes are:
   A. Isobaric Mode.
   B. Auto Mode.
   C. Manual Mode.
   D. A and B above.

9. If the #1 MADC fails in flight:
   A. The controller amber light illuminates and switches to FL Isobaric Mode.
   B. It remains in the CA AUTO Mode.
   C. Only manual control remains
   D. It automatically switches to the EXER Mode.

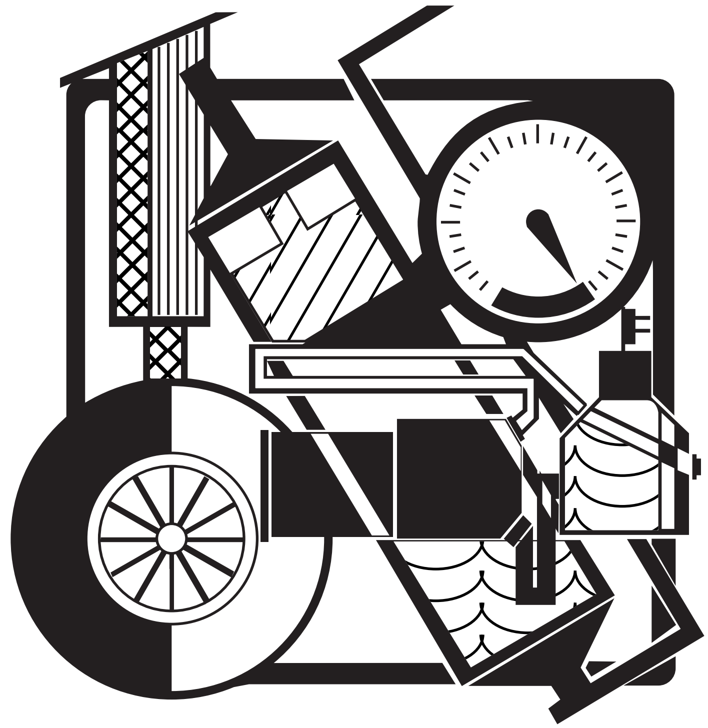
10. During preflight, the controller is normally set to:
    A. Destination field elevation.
    B. Cruise plus 1000 feet in the FL mode.
    C. Field pressure altitude plus 500 feet.
    D. 300 feet to 500 fpm on the cabin rate of climb control.

CHAPTER 13
HYDRAULIC POWER SYSTEMS

CONTENTS

ILLUSTRATIONS

CHAPTER 13
HYDRAULIC POWER SYSTEMS



INTRODUCTION

The Citation EXCEL utilizes two engine-driven hydraulic pumps, one on each engine, to provide pressure for the hydraulic system. The system provides operating pressure for five sub-systems: landing gear, speed brakes, flaps, two-position horizontal stabilizer, and thrust reversers.

GENERAL

The main hydraulic system is an "open center" type system utilizing a phosphate ester base hydraulic fluid (Skydrol or Hyjet). Pressure is supplied by two constant displacement, engine-driven pumps while the engines are running. A bootstrap type reservoir stores fluid and provides positive pressure to the pump inlets. All fluid is routed through system pressure and return filters. Hydraulic fluid continuously flows through the system at low pressure (normal resistance pressure) until a sub-system is selected. When a sub-system is selected, the system control valve closes providing pressure to operate that system. A system relief valve prevents system overpressure by dumping excess fluid to the reservoir. Check valves separate the left and right pumps so that either pump can pressurize the main hydraulic system.

Separate solenoid valves provide pressure to various sub-systems as selected by the flight crew, i.e., landing gear, flaps/horizontal stabilizer, speed brakes, and thrust reversers.

Various lights located in the cockpit provide warning annunciations and position indicators.

MAJOR COMPONENTS

RESERVOIR

The reservoir is mounted in the tailcone area (Figure 13-1).

A visual fluid level indicator on the aft end indicates FULL when properly serviced with 215 cubic inches of fluid, OVERFULL at 360 cubic inches, or REFILL at 175 cubic inches (Figure 13-1).

If the fluid level drops to 74 cubic inches, the amber **LO HYD LEVEL** annunciator illuminates. Checking reservoir fluid level is a preflight inspection item.

A relief valve on top of the reservoir opens to prevent over pressure in case the reservoir is over serviced. It may also be manually opened for bleeding the system or releasing excess fluid.

When the hydraulic system is not under pressure, a spring-loaded piston in the reservoir maintains pressure on the hydraulic fluid to provide head pressure to the pumps, and absorb pressure spikes, thus eliminating the need for a main system accumulator.

When the main system is pressurized (control valve closed), additional pressure is applied to the spring-loaded piston to maintain adequate pressure to the pumps while under demand to prevent cavitation.

HYDRAULIC SERVICE PANEL

An access door on the right side of the fuselage behind the right wing allows access to the service panel (Figure 13-2). Servicing the reservoir requires pressurizing equipment such as a hydraulic service cart or hand-operated pump.

DRAIN MAST

A hydraulic drain mast is mounted on the bottom of the fuselage slightly forward of the service access door lower hinge (Figure 13-2). Excess fluid from the reservoir may drain through the relief valve to the drain mast. The mast is extended to prevent caustic hydraulic fluid from spraying on exterior paint.

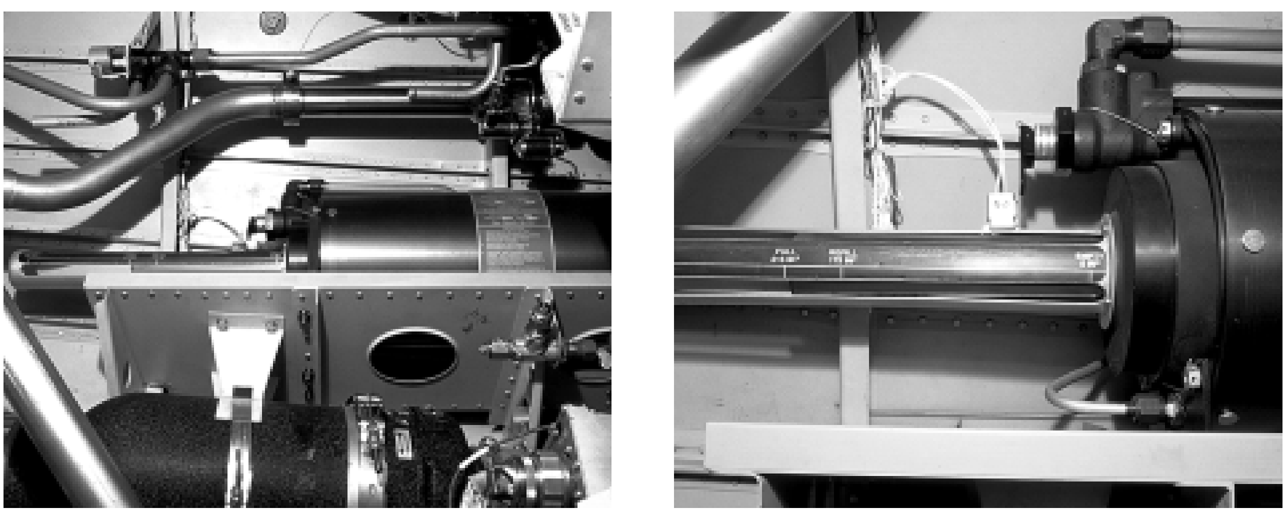


Figure 13-1 Hydraulic Reservoir

Figure 13-2 Hydraulic Service Connections/Drain Mast

PUMPS

Two constant-volume displacement gear pumps, driven by the accessory section of each engine, circulate fluid through the system when the engines are operating. Either pump is capable of generating 1,500 psi and operating all subsystems.

SYSTEM CONTROL VALVE

A solenoid-operated system control valve is the "heart" of the system. It is spring-loaded "open" to route pump output to the return line. It is energized closed by selecting operation of a sub-system, and 1,500 psi hydraulic pressure is produced (Figure 13-3). A mechanical

relief valve in parallel with the control valve maintains system pressure at approximately 1,500 psi. If electrical power is interrupted, the valve fails "open." The control valve is powered from the emergency DC bus system through the HYD CONTROL CB on the pilot's CB panel.

FIREWALL SHUTOFF VALVES

Individual hydraulic firewall shutoff valves are installed in the supply line to each hydraulic pump. The valves are electric motor operated and are controlled by ENG FIRE switchlights on the glareshield. The valves are normally open and selected closed only in the event of an engine fire or to perform maintenance. The valves require main DC power to operate closed or open (Chapter 8, FIRE PROTECTION).

FILTERS

The system incorporates three fluid filters, two for filtering fluid leaving the pumps and one for filtering return fluid prior to entering the reservoir. Each filter incorporates a bypass valve that opens if the filter element clogs. There is no cockpit indication of a filter bypassing.

FLOW SWITCHES/CHECK VALVES

A flow switch in each pump pressure line controls the **LO HYD FLOW L/H** annunciators (Figure 13-3). As flow from a pump increases to 1.33 Gallons Per Minute (GPM)/503 Liters Per Minute (LPM) flow rate, the annunciators will extinguish. Decreasing flow rates of 0.35 to 0.55 GPM/1.32 to 2.08 LPM will illuminate the **LO HYD FLOW L/H** annunciators. The annunciators will illuminate "steady" on the ground with the engines shutdown prior to starts.

Check valves installed in the pump pressure lines prevent backflow into the pumps. They also ensure that a defective pump is isolated from the system to allow the functioning pump to supply system pressure, i.e., sheared pump shaft, pump leaking, engine shutdown, etc.

HYDRAULIC SYSTEM OPERATION

GENERAL

When an engine is started, the hydraulic pump draws fluid from the reservoir through the normally open firewall shutoff valve. Pump output flow, through the flow switch, opens a circuit to extinguish the L or R segment of the **LO HYD FLOW** annunciator light. Assuming that a sub-system is not activated, the de-energized system control valve is open, bypassing pump output to return. As the second engine is started, the entire **LO HYD FLOW** annunciator is extinguished (Figure 13-3).

When operation of any sub-system is initiated, a circuit is completed to energize the system control valve closed. As pressure increases, the **HYD PRESS** annunciator illuminates. System pressure is limited to 1,500 psi by the system relief valve. When the selected operation is completed, the valve spring-loads to the "open" position, bypassing pump output to return. The system depressurizes, and the **HYD PRESS** annunciator extinguishes. The system remains in the open center condition until another sub-system is selected for operation.

When an engine is shut down, the applicable segment of the **LO HYD FLOW** annunciator illuminates. With both engines shut down, the entire annunciator illuminates. Loss of pump output during system operation is indicated by illumination of the L or R segment, as applicable.

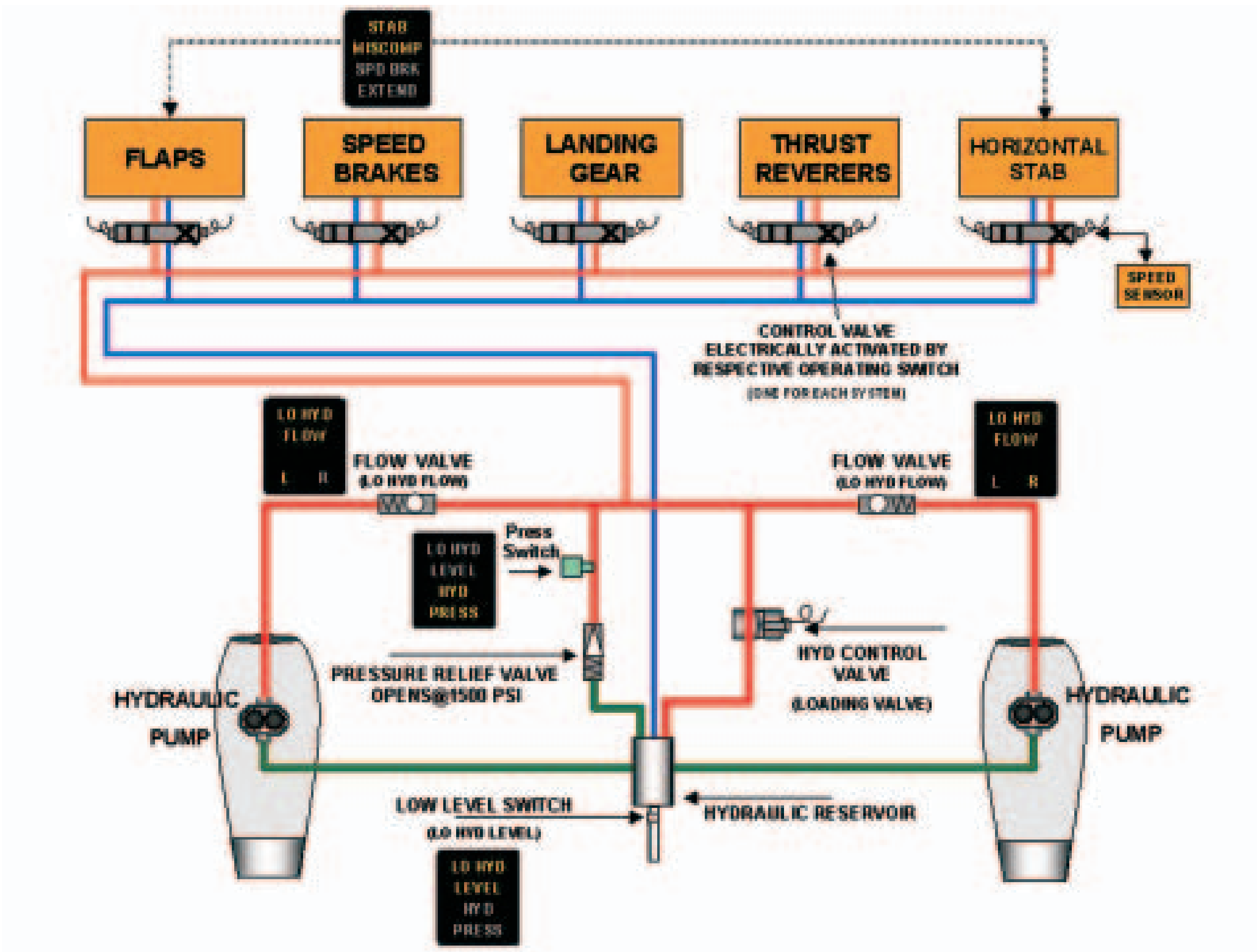


Figure 13-3 Hydraulic System

NORMAL OPERATION

After one or both engines are operating, the hydraulic pumps are circulating hydraulic fluid through the “open center” system at low resistance pressure and the associated **LO HYD FLOW L/R** annunciators are extinguished. When a sub-system is selected, i.e., landing gear, flaps/horizontal stabilizer, speed brakes, or thrust reversers, the system control valve will close and as pressure increases the **HYD PRESS** annunciator will illuminate “steady”. After a sub-system has cycled to its proper position, except the thrust reversers deployed, the **HYD PRESS** light will extinguish

NOTE

The **HYD PRESS** annunciator will remain illuminated while the thrust reversers are deployed, See THRUST REVERSERS later in this Chapter.

NOTE

The **HYD PRESS** annunciator will initially illuminate “steady” for 40 seconds. If after 40 seconds have elapsed and the **HYD PRESS** annunciator remains illuminated, it will commence “flashing” and trip the MASTER CAUTION lights steady.

ABNORMAL OPERATION

Low Hydraulic Flow (LO HYD FLOW L or R Caution Light On

If a **LO HYD FLOW L** or **R** annunciator illuminates “flashing”, it indicates an inoperative hydraulic pump. The system will continue to operate normally with one pump operational.

If both **LO HYD FLOW L** and **R** annunciators illuminate. The hydraulic system is inoperative and the airplane should be landed as soon as practical.

Plan the approach and landing without the use of speed brakes, flaps and the horizontal stabilizer.

NOTE

If the flaps are in the approach or landing position when the hydraulic system failed, moving the flap handle my cause the flaps to float up to a trail position.

Plan to lower the landing gear by the emergency extension method (refer to the ABNORMAL PROCEDURES section in the FSI PTM Volume 1). The thrust reversers will not be available after touchdown.

Hydraulic System Remains Pressurized (HYD PRESS Caution Light Remains On After System Cycle is Completed)

Indicates the Hydraulic System Control valve did not open. Pulling the HYD CONTROL CB on the pilot’s CB panel should de-energize the control valve open and the system should depressurize and the caution light will extinguish.

If system depressurizes

Resetting the HYD CONTROL CB prior to approach and landing will allow the sub-systems to be operational during this important phase of the flight. If the system remains pressurized after resetting the CB, it is not necessary to pull the CB. The short time the system is pressurized during the approach and landing phase will not damage the hydraulic system.

If system remains pressurized

Indicates the system control valve has failed in the closed position. The HYD CONTROL CB should be reset, maintain airspeed 200 KIAS or below and do not exceed FL310.

NOTE

Skydrol and Hyjet hydraulic fluids build up excessive heat under pressure.

Observing the airspeed and altitude recommendations will help prevent fluid temperatures from becoming dangerous. Reducing engine RPM to maintain 200 KIAS lowers engine driven hydraulic pumps speed, and better ambient cooling is obtained at the lower altitudes where the air is more dense.

The aircraft should be landed as soon as practical. All sub-systems will remain operational for approach and landing.

Low Hydraulic Fluid Level (LO HYD LEVEL Caution Light On)

On the ground

Maintenance should be performed and/or serviced prior to flight.

In flight

The sub-systems may not operate. The aircraft should be landed as soon as practical. Use care during approach and landing (refer to ABNORMAL PROCEDURES section in the FSI PTM Volume 1).

THRUST REVERSERS

DESCRIPTION

The Citation EXCEL is equipped with hydraulically operated, electrically controlled, target-type, vertically oriented doors or buckets, and when deployed, direct exhaust gases forward to provide a deceleration force during landing roll.

The thrust reverser buckets are attached to the aft flange of each engine and employ a 4 bar link configuration with dual overcenter links. The thrust reverser buckets are designed to remain stowed in the event of loss of power (electrical or hydraulic) through incorporation of an over-center (mechanical) load in the actuation system.

When deployed, the reversers are maintained in position by hydraulic pressure.

During normal operation, hydraulic pressure is isolated when the reversers are stowed. They are maintained in the stowed position by the overcenter condition of the operating bar mechanisms.

PROTECTION

A solenoid lock in the throttle quadrant prevents increasing engine reverse thrust until the associated reverser reaches the fully deployed position.

A throttle feedback system moves the Fuel Control Unit (FCU) lever and throttle to idle if the reversers deploy inadvertently.

Thrust reverser operation is limited to ground operations only. The control circuitry is wired through both left and right main gear squat switches.

NOTE

Either landing gear squat switch that senses a Ground-On-Ground (GOG) condition, will provide a ground to allow either or both thrust reversers to be deployed individually or simultaneously.

CONTROL

Reverser levers, piggyback-mounted on the throttles, control the thrust reversers. Each reverser lever has three positions - forward or stow, a detented reverse idle (deploy) position, and a full aft reverse thrust “stop” position. Full reverse thrust should be limited to 75% of takeoff thrust.

When a reverser lever is moved to the reverse idle (deploy) position, the solenoid lock (mentioned earlier) will prevent further aft movement until the reverser is fully deployed.

A microswitch in the throttle quadrant provides for electrical control. The switch is closed when the reverser lever is moved from the stow position, applying power to (1) close the hydraulic control valve and pressurize the hydraulic system, (2) open the thrust reverser isolation valve directing pressure to the reverser hydraulic system, and (3) energize the reverser control valve to deploy the buckets, provided a ground is supplied by either or both squat switches (Figure 13-4).

Electrical power for the thrust reversers is provided by main DC power through CBs located on the pilot’s CB panel.

INDICATIONS

Each reverser has three lights on the glareshield panel — **ARM, UNLOCK,** and **DEPLOY**. The amber ARM light circuit is completed by a pressure switch indicating the isolation valve is open and system pressure is available to the reverser control valve.

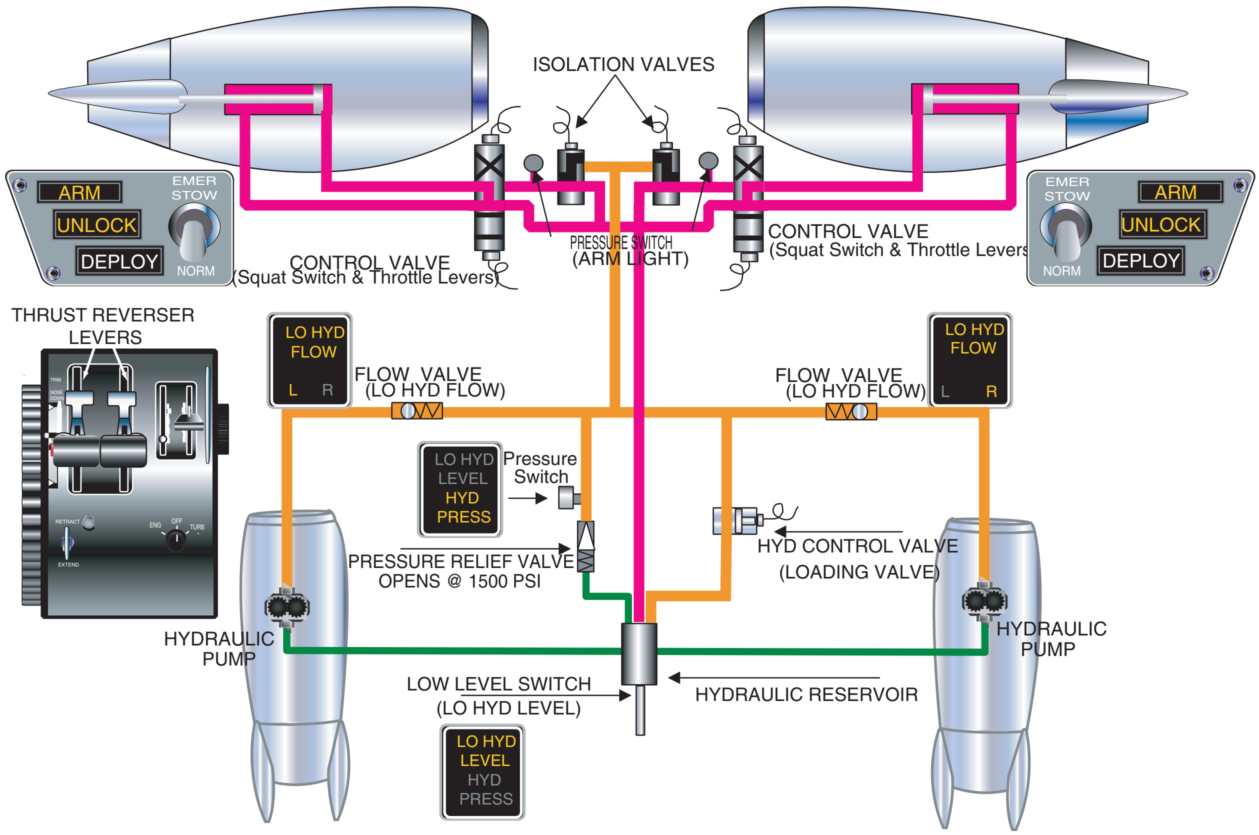


Figure 13-4 Thrust Reverser System

In addition to the three reverser lights, a light on the annunciator panel labeled "**HYD PRESS**" will illuminate to indicate that the hydraulic control valve is closed and the hydraulic system is pressurized.

The amber **UNLOCK** light circuit is completed by a limit switch which closes when the reverser mechanism initially moves from the mechanically locked stow position.

The white **DEPLOY** light indicates the reverser door mechanism has reached the fully deployed limit switch position.

OPERATION

Raising the piggyback thrust reverser levers with the airplane on the ground and the throttles at IDLE, will actuate the reverser(s). The **HYD PRESS** annunciator will illuminate steady followed immediately by "steady" **ARM** light(s) illuminating as the thrust reverser isolation valve(s) energize open. Next, the control valve(s) energize open, provided a squat switch is in the GOG mode, and allows hydraulic pressure to the thrust reverser actuators mounted on the engine nacelle(s). The overcenter locks are released followed immediately by the amber **UNLOCK** light(s) illuminating "steady". As the thrust reverser buckets are driven to the full deploy position, the white **DEPLOY** light(s) illuminate (Figure 13-4).

The reverser lever solenoid locks will release. The reverser levers may now be moved aft to accelerate the engines if so desired. This last movement acts only on the FCU to increase reverse thrust. The engine throttles are held in idle by a mechanical interlock in the pedestal.

While reverse thrust is maintained, the **ARM, UNLOCK, DEPLOY** and **HYD PRESS** lights remain illuminated.

Stow the reverser(s), by moving the reverser lever(s) full forward and down. This action energizes the reverser control valve to the stow position, and directs hydraulic pressure to the stow side of the reverser actuators. The **DEPLOY** light will extinguish, followed almost immediately by the **UNLOCK, ARM**, and **HYD PRESS** lights extinguishing, indicating the reverser doors are in the fully stowed position.

CAUTION

DO NOT ATTEMPT TO MOVE THE ENGINE THROTTLES FORWARD BEFORE THE UNLOCK LIGHTS GO OUT.

THE THROTTLE FEEDBACK MECHANISM THAT RETARDS THE THROTTLE TO IDLE MUST NOT BE RESISTED BY A FRICTION KNOB SETTING THAT IS TOO TIGHT OR BY THE PILOT'S HAND. IT COULD RESULT IN THROTTLE CABLE SYSTEM DAMAGE.

CAUTION

DEPLOYMENT OF THE THRUST REVERSERS, ESPECIALLY AT HIGHER-THAN-NORMAL LANDING SPEEDS, CAUSES A NOSE UP PITCHING MOMENT, WHICH MUST BE COUNTERED BY FORWARD PRESSURE ON THE CONTROL YOKE. IF NOT COUNTERED, THIS COULD LEAD TO A PORPOISE AND POSSIBLE NOSE STRUT DAMAGE.

WARNING

IF THE REVERSER IS INADVERTENTLY DEPLOYED, THE THROTTLE MUST NOT BE MOVED FORWARD UNTIL THE REVERSER IS FULLY STOWED. OVERRIDING THE FEEDBACK SYSTEM WHILE THE REVERSER IS DEPLOYED COULD RESULT IN A DANGEROUS ASYMMETRICAL THRUST CONDITION AND LOSS OF PERFORMANCE.

EMERGENCY STOW

An emergency stow system is incorporated which bypasses the normal sequencing system. This system is used in case of an inadvertent deployment in flight or if the normal stow system fails.

A two-position switch for each reverser is located inboard of the reverser lights. The switch is labeled “STOW EMER - NORM.” Placing a STOW switch to the EMER position will close the hydraulic system control valve and cause the reverser isolation and control valves to energize to the stow position (Figure 13-4). The **HYD PRESS** light will illuminate and the **DEPLOY** and **UNLOCK** lights will extinguish. The **HYD PRESS** light and the **ARM** light will remain on continuously in the STOW position, thus holding the reversers STOWED by applying continuous hydraulic pressure (mechanical, overcenter stow locks may be inoperable).

The emergency stow system is tested before each flight following a normal deploy cycle.

If either an **ARM** or **UNLOCK** light illuminates in flight, the master warning lights will also flash.

The LH thrust reverser uses left main DC power from the LH thrust reverser circuit breaker for normal STOW-DEPLOY operation. However, it uses power from right main DC system through the RH thrust reverser circuit breaker for emergency stow. Conversely, the RH thrust reverser RH CB provides power for normal RH thrust reverser operation and power from the LH thrust reverser CB for emergency stow.

NOTE

If a fire switchlight is pushed for an engine fire, the reverser isolation valve remains de-engergized closed and the affected engine reverser is prevented from deploying inadvertently or intentionally.

NORMAL OPERATIONS

After landing, when the throttles are at idle and the nose wheel is on the ground, raise the thrust reverser levers to the idle deploy detent. The thrust reversers are more effective at higher speeds. The **HYD PRESS** and **ARM** lights will illuminate, followed almost immediately by the **UNLOCK** lights, and then by the **DEPLOY** lights. The reverser lever solenoid locks will release. The reverser levers may now be moved aft to accelerate the engines if so desired. This last movement acts only on the FCU to increase reverse thrust. The engine throttles are held in idle by mechanical interlocks in the pedestal. While reverse thrust is maintained, the **ARM, UNLOCK, DEPLOY** and the **HYD PRESS** lights will remain illuminated.

As the airplane decelerates toward 60 KIAS, reverse thrust should be decreased to achieve idle reverse power at 60 KIAS. The thrust reverser indicating lights and the **HYD PRESS** light will all remain on. Reverse idle may be maintained to assist further deceleration by drag and attenuation of thrust.

The reversers are stowed by placing the T/R levers full forward. The **DEPLOY, UNLOCK, ARM** and **HYD PRESS** lights will extinguish in that order as the reversers stow.

Rotary Test

Positioning the Rotary Test switch to T/REV, should cause the amber **ARM, UNLOCK** and white **DEPLOY** lights to illuminate on both sides of the annunciator panel and trip both MASTER WARNING RESET lights "flashing". The test ensures that if an **ARM** or **UNLOCK** light illuminates in flight the MASTER WARNING lights will provide immediate warning.

Thrust Reversers Preflight Check

During taxi while conducting the TAXI checklist, the thrust reversers are checked as follows:

1. Deploy both reversers and check sequencing and timing of lights. The amber **ARM** lights should illuminate simultaneously with the **HYD PRESS** annunciator. Without delay, the amber **UNLOCK** lights followed by the white **DEPLOY** lights should illuminate.

   NOTE

   AB configuration aircraft with rudder bias: Verify that the **RUDDER BIAS** caution annunciator remains extinguished.

2. Select EMER STOW, both switches, and check sequencing and timing of lights. The **DEPLOY** and **UNLOCK** lights should extinguish in that order. **ARM** lights and **HYD PRESS** annunciator should remain illuminated.

3. Stow the thrust reverser levers (full forward). The **ARM** lights and **HYD PRESS** annunciator remain illuminated.

4. Select EMER STOW switch, NORM, and verify all thrust reverser lights extinguished. The **ARM** lights and the **HYD PRESS** annunciator should extinguish in that order as both EMER STOW switches are placed in NORM.

CAUTION

DO NOT TAKE OFF WITH THE THRUST REVERSER EMER STOW SWITCHES IN "EMER". THE REVERSERS SHOULD BE AVAILABLE IN CASE AN EMERGENCY ABORT IS REQUIRED.

WITH THE EMERGENCY STOW SWITCHES IN "EMER", THE HYDRAULIC SYSTEM WOULD REMAIN PRESSURIZED TO MAINTAIN THE THRUST REVERSERS STOWED AND POSSIBLY PREVENT THE GEAR FROM RETRACTING PROPERLY AND PREVENTING THE FLAPS AND HORIZONTAL STABILIZER FROM OPERATING PROPERLY.

EMERGENCY/ABNORMAL OPERATIONS

Thrust Reverser Inadvertent Deployment During Takeoff

Speed Below V_1 — Takeoff Should Be Aborted

Apply wheel brakes as required (maximum braking if needed). Reduce throttles to idle.

NOTE

The throttle on the affected engine(s) should have retarded automatically to idle as the thrust reverser(s) inadvertently deployed.

NOTE

AB configuration - During inadvertent thrust reverser deployment on one engine, Rudder Bias will initially move rudder toward opposite engine.

The speed brakes should be extended and deploy the unaffected thrust reverser (AB configuration - Rudder Bias disabled). Check the thrust reverser indicator lights illuminated, **ARM, UNLOCK** and **DEPLOY** lights. Apply reverse thrust as needed to aid in stopping the airplane.

Speed Above V_1 — Takeoff Should Normally Be Continued.

Place the affected engine EMER STOW switch to EMER (AB configuration - Rudder Bias disabled) and check the affected engine throttle at IDLE (should have automatically retarded to IDLE upon reverser deployment). Continue the takeoff and retract the gear after a positive rate of climb is established. Do not exceed 140 KIAS until the reverser stows. Continue the climb at V_2 until clear of obstacles.

After the thrust reverser stows, the affected engine throttle may be advanced to adjust power as required.

After all obstacles are cleared, the airplane may be accelerated to V_2 +10 KIAS minimum and the flaps may be retracted as required. After the thrust reverser stows, airspeed may be increased, if desired, not to exceed 200 KIAS. Restrict altitude to FL310 and below. The airspeed and altitude restriction is to prevent hydraulic fluid temperature from increasing to dangerous levels. Maintaining airspeed not to exceed 200 KIAS should also prevent the reverser from deploying again. The airplane should be landed as soon as practical.

If Thrust Reverser Will Not Stow

Check both thrust reverser CBs "in" on the pilot's CB panel. Both T/R CBs "in" should provide normal and/or emergency stow capability for the reversers. If the CBs check "in" and the affected reverser still will not stow, shutdown the engine and maintain an airspeed of 140 KIAS or below.

The airplane should be landed as soon as practical, refer to the EMERGENCY PROCEDURES section in the FSI PTM Volume 1.

NOTE

The BEFORE LANDING (WITH THRUST REVERSER DEPLOYED) emergency procedure in PTM Volume 1, stipulates landing with flaps at TAKEOFF and APPROACH position, 15°, and airspeed VAPP minimum. Multiply charted landing distances by 1.4 for flaps 15°.

WARNING

DO NOT INITIATE GO-AROUND BELOW 600 FEET AGL WITH A THRUST REVERSER DEPLOYED.

Thrust Reverser Inadvertent In-Flight Deployment

If a thrust reverser should inadvertently deploy inflight, corrective procedures are essentially the same as experiencing an inadvertent T/R deployment during takeoff after V_1. The airplane will immediately develop an adverse yaw and pitch up tendency which can be corrected by taking manual control of the aircraft and disengaging the autopilot, if engaged.

Place the EMER STOW switch to EMER, check the affected engine throttle at IDLE, and reduce airspeed to 140 KIAS or below.

The thrust reverser indicator lights should be checked. If emergency stow operates properly, the **UNLOCK** and **DEPLOY** lights should be extinguished, and the **ARM** light and **HYD PRESS** annunciator should be illuminated. The affected engine throttle may be matched with the other engine. Restrict airspeed and altitude not to exceed 200 KIAS and FL310 respectively. Land as soon as practical.

If Thrust Reverser Will Not Stow.

Check the L and R THRUST REVERSER CBs in on the pilot's CB panel.

If Thrust Reverser Still Will Not Stow

Place the affected engine throttle to CUT OFF, maintain airspeed not to exceed 140 KIAS, and land the airplane as soon as possible. Refer to the EMERGENCY PROCEDURES section in the FSI PTM Volume 1.

Thrust Reverser Unlock Light On In Flight

If a thrust reverser UNLOCK light should illuminate in flight, activate the EMER STOW switch to EMER. Check the T/R levers stowed, full forward position.

If Light Will Not Extinguish

Check the L and R THRUST REVERSER CBs "in" on the pilot's CB panel. Maintain airspeed not to exceed 200 KIAS and restrict altitude to FL310 maximum. Land as soon as practical.

Thrust Reverser Arm Light On In Flight

If a T/R amber ARM light illuminates in flight, check the thrust reverser levers stowed, full forward and verify the EMER STOW switch is in the NORM position.

If ARM Light is Still Illuminated

Check for illumination of the HYD PRESS annunciator.

If HYD PRESS Caution Light is Not Illuminated

Land the airplane as soon as practical.

If HYD PRESS Caution Light is Illuminated (Indicates the T/R Isolation Valve is Open)

Place affected EMER STOW switch to EMER, maintain airspeed 200 KIAS or below, and restrict altitude not to exceed FL310. Land the airplane as soon as practical.

OTHER HYDRAULIC SUB-SYSTEMS

Other main system hydraulic powered subsystems include landing gear, speedbrakes, flaps, and the horizontal stabilizer.

Application of hydraulic power to these subsystems is presented in Chapter 14, Landing Gear and Brakes, and in Chapter 15, Flight Controls.

LIMITATIONS

HYDRAULIC FLUID

Use Skydrol 500A, B, B-4, C, or LD-4; or Hyjet, Hyjet W, III, IV, IVA, or IVA Plus only.

THRUST REVERSERS

Reverse thrust power must be reduced to the idle reverse detent position at 60 KIAS on landing roll.

Maximum reverse thrust setting is limited to 75% of takeoff thrust.

Maximum allowable thrust reverser deployed time is 3 minutes in any 10 minute period.

Engine static ground operation is limited to idle power (if thrust reversers are deployed).

Use of thrust reversers is prohibited during touch and go landings.

The thrust reverser(s) must be verified to be operational by the Taxi test in Section III, Normal Procedures, in the *Airplane Flight Manual (AFM)*.

The use of thrust reversers to back the airplane is prohibited.

QUESTIONS

1. The system control valve is:
   A. Spring-loaded closed.
   B. Spring-loaded open.
   C. Energized closed.
   D. B and C.

2. Depressing an ENG FIRE switch light:
   A. Shuts off hydraulic fluid to the pump.
   B. Trips the generator field relay.
   C. Arms the fire extinguishing system.
   D. All of the above.

3. Closing a hydraulic firewall shutoff valve is indicated by:
   A. A warning horn.
   B. Illumination of the applicable segments of the **F/W SHUTOFF** annunciator and **LO HYD FLOW** annunciator.
   C. Illumination of the **HYD PRESS** annunciator.
   D. None of the above.

4. If DC power is lost to the hydraulic system control valve:
   A. Fails to the closed position.
   B. Is not affected.
   C. Fails to the open position.
   D. None of the above.

5. The hydraulic system provides pressure to operate the:
   A. Landing gear and speedbrakes only.
   B. Antiskid brakes, landing gear, and flaps.
   C. Speedbrakes, landing gear, thrust reversers, horizontal stabilizer, and flaps.
   D. Speedbrakes, landing gear, and wheel brakes.

6. The reservoir quantity indicator is located:
   A. In the right forward baggage compartment.
   B. On the copilot's instrument panel.
   C. On the right engine near the oil filter.
   D. In the tail cone area.

7. Low reservoir fluid level is indicated by illumination of the:
   A. **LO HYD LEVEL** annunciator.
   B. **HYD PRESS** annunciator.
   C. **L/R LO HYD LEVEL** annunciator.
   D. **L/R LO HYD FLOW** annunciator.

8. Hydraulic system operation is indicated by illumination of the:
   A. **LO HYD LEVEL** annunciator.
   B. **HYD PRESS** annunciator.
   C. **L/R LO HYD LEVEL** annunciator.
   D. **L/R LO HYD FLOW** annunciator.

9. The correct statement concerning the hydraulic system is:
   A. The **HYD PRESS** annunciator illuminates anytime an engine-driven pump is operating.
   B. The **HYD PRESS** annunciator illuminating while the gear is extending may indicate a failed hydraulic pump.
   C. The **LO HYD FLOW L/R** annunciator illuminates whenever reservoir fluid level low.
   D. A **L** or **R LO HYD FLOW** annunciator illuminating may indicate a failed hydraulic pump.

10. The thrust reversers:
    - A. May be deployed only when the throttles are in IDLE.
    - B. Must have both EMER STOW switches in EMER for takeoffs to guard against inadvertent deployment during that critical phase of flight.
    - C. May be left in idle reverse until the airplane is brought to a full stop.
    - D. Both A and C.

11. When normal deployment of the thrust reversers is obtained, the following annunciator lights should be illuminated:
    - A. **ARM, UNLOCK, DEPLOY**.
    - B. **DOOR NOT LOCKED, ARM, UNLOCK, DEPLOY.**
    - C. **HYD PRESS, ARM, UNLOCK, DEPLOY.**
    - D. **DOOR NOT LOCKED, HYD PRESS, DEPLOY.**

12. The incorrect statement regarding the use of thrust reversers is:
    - A. They may be used in flight to slow the airplane.
    - B. They should not be used on touch-and-go landings.
    - C. The reversers must be in idle reverse by 60 KIAS.
    - D. Either squat switch on the ground will allow both reversers to deploy.

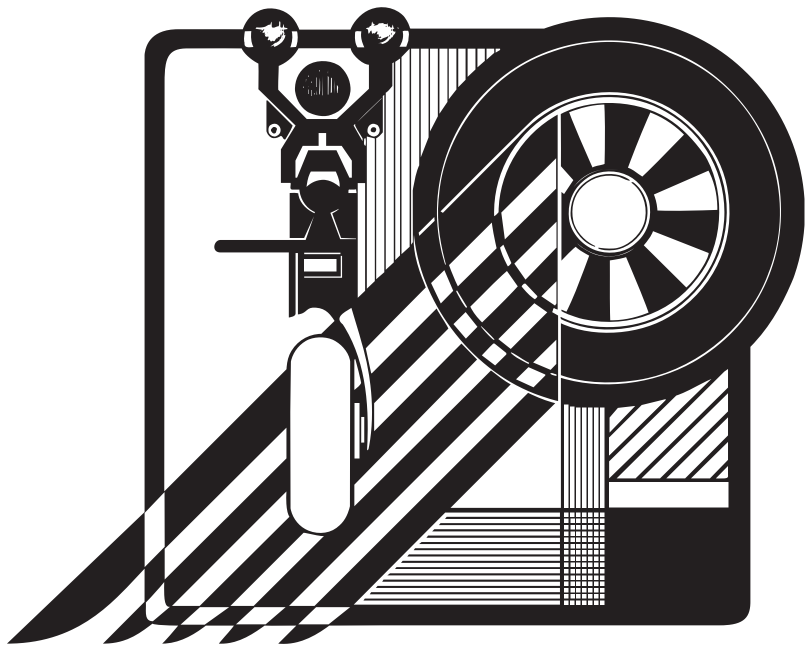
13. The master warning lights:
    - A. Have nothing to do with the reverser system.
    - B. Will illuminate if an ARM light illuminates in flight.
    - C. Will illuminate if the **HYD PRESS** light remains illuminated after the **DEPLOY** light is illuminated on the ground.
    - D. Will not illuminate if an **UNLOCK** light illuminates in flight.

CHAPTER 14
LANDING GEAR AND BRAKES

CONTENTS

ILLUSTRATIONS

CHAPTER 14
LANDING GEAR AND BRAKES



INTRODUCTION

The EXCEL landing gear is electrically controlled and hydraulically actuated. Gear position and warning are provided by green and red indicator lights and an aural warning.

Nose wheel steering is mechanically actuated through linkage from the rudder pedals. A self-contained shimmy damper is located on top of the nose gear strut.

An antiskid power brake system is standard. A backup emergency pneumatic braking system is incorporated.

GENERAL

The EXCEL features a tricycle gear, with fuselage mounted, single wheel telescoping nose gear and two wing mounted, single wheel, trailing link main gear.

Gear warning and position indication is provided by one red and three green indicator lights, and aural warning if the throttles or flaps, and gear position are not compatible.

Nose wheel steering is mechanically actuated by cable linkage from the rudder pedals. The system is enabled with the gear extended, in flight or on the ground. Nose gear centering is accomplished mechanically during retraction.

A power brake/antiskid system initiated by rudder pedal toe brakes is incorporated. It uses a separate hydraulic system that is isolated from the main hydraulic system. Each main gear wheel houses a multiple carbon disc brake assembly that is actuated by hydraulic pressure from a dedicated electrical pump, or stored nitrogen pressure during emergency braking.

A mechanical actuated parking brake is provided for parking the aircraft.

LANDING GEAR

DESCRIPTION

The main and nose landing gear struts are conventional oil-over-air struts. The landing gear is normally hydraulically actuated but can be mechanically and pneumatically released and extended if normal gear extension fails. The gear can be extended at airspeeds up to 250 KIAS and retracted at airspeeds up to 200 KIAS, and operated with the gear extended at airspeeds up to 250 KIAS. It takes approximately six seconds to extend or retract the landing gear.

Main Gear

The main gear assembly (Figure 14-1) includes a trunion, oleo struts, actuators, trailing link, a single wheel with a multiple disc brake, and a squat switch that senses in-flight/on-ground conditions.

The main gear is locked in the retracted position by uplock hook mechanisms. Prior to extension, an uplock hydraulic actuator must release each uplock before hydraulic pressure can reach the main actuators to extend the gear. When the gear is fully extended, internal locks within the gear actuators engage. The down lock mechanisms consists of a spring loaded locking ring held in a groove on the actuator piston. It can be released only with hydraulic pressure applied to the retract side of the actuator. External down lock pins are not required.

Figure 14-1 Main Landing Gear

The main gear doors are actuated mechanically by gear movement. They cover only the main gear struts when retracted. The wheel assembly fairs into the wheel well and is not covered.

To prevent injury to personnel, each main gear wheel incorporates a fusible plug that melts to deflate the tire if excessive temperature is generated by an overheated brake

Nose Gear

The nose gear assembly (Figure 14-2) includes a strut, hydraulic actuators, torque links, a single wheel, and a self-contained shimmy damper. The nose gear is held in the retracted position by an uplock hook mechanism that is released by an uplock hydraulic actuator prior to gear extension. When the gear is extended, an internal locking mechanism in the gear actuator engages to lock the gear down. This locking device is similar to the main gear actuator locks. An external down lock pin is not required for the nose gear. The nose gear is mechanically centered during retraction.

Three doors are actuated mechanically by nose gear movement to completely enclose the nose gear and wheel at retraction. All three gear doors open as the gear extends and remain open with the nose gear extended.

Figure 14-2 Nose Wheel Assembly

CONTROLS AND INDICATORS

The landing gear control handle may be installed on the pilot's or copilot's lower instrument panel (operator's option, Figure 14-3). Gear position is indicated by one red and three green indicator lights on the gear control panel. A warning horn provides aural warning of abnormal conditions.

NOTE

The Citation EXCEL is not equipped with a gear warning horn silence button.

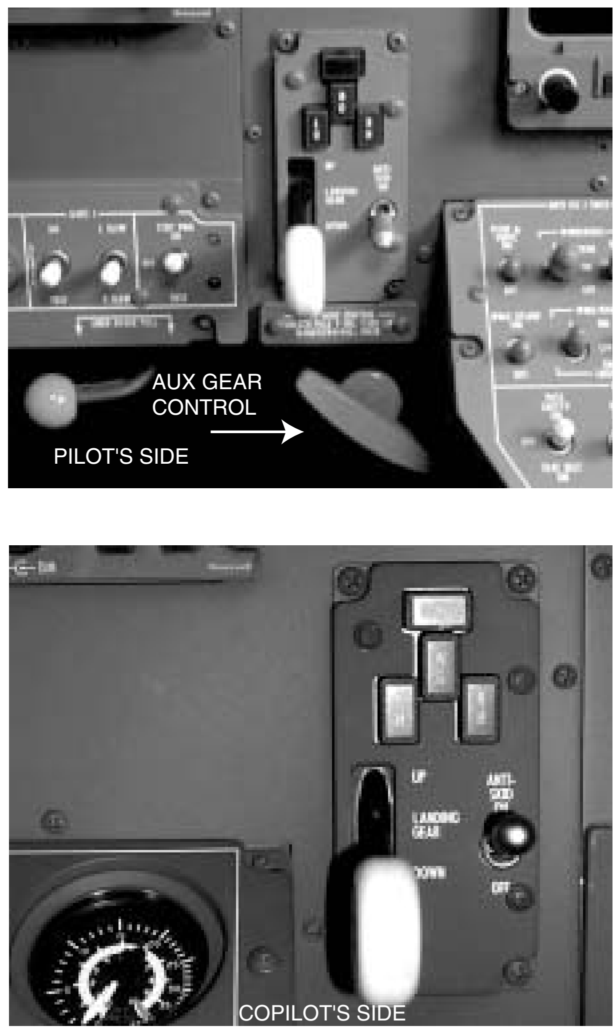


Figure 14-3 Landing Gear Control Panel

Controls

Landing gear control is powered through the emergency DC bus GEAR CONTROL CB located on the LH circuit breaker panel. Raising or lowering the LDG GEAR handle actuates switches to complete circuits to open the extend or retract ports of the gear solenoid control valve.

On the ground (LH squat switch, Figure 14-4), a spring-loaded plunger holds the handle in the DOWN position, preventing inadvertent movement of the handle to the UP position.

Figure 14-4 Squat Switch

CAUTION

NEVER ATTEMPT TO PULL THE GEAR HANDLE UP DURING TAXI. SPECIAL ATTENTION MUST BE GIVEN TO CHECKING THE GEAR HANDLE IN THE DOWN POSITION BEFORE DEPRESSING AN ENGINE START BUTTON TO PREVENT INADVERTENT NOSE GEAR RETRACTION AS HYDRAULIC PRESSURE IS GENERATED.

DC power for the gear position indicator lights, aural warning, and the locking solenoid on the gear handle is through the emergency bus GEAR WARNING circuit breaker on the left circuit breaker panel. The GEAR warning circuit breaker should not be confused with the GEAR CONTROL circuit breaker in the same SYSTEMS section of the CB panel.

In flight, with the left main gear squat switch in the airborne mode, the locking solenoid is energized to retract the plunger and free the handle for movement to the UP position. This safety feature cannot be overridden. If the solenoid fails or electrical power is lost, the gear handle cannot be moved to the UP position. The gear handle must be pulled out of the detent prior to movement UP or DOWN.

Indicators

The green NOSE, LH, and RH lights on the gear control panel indicate gear down and locked. As each gear locks down, its respective green light is illuminated.

The red GEAR UNLOCK light indicates an unsafe gear condition. It illuminates when the gear handle is moved out of the UP detent and remains on until all three gear are down and locked. During retraction, the light comes on when the first down lock is released and remains on until all three gear are up and locked.

Normal indication with the gear down is three green lights illuminated and the red UNLOCK light extinguished. All lights should be out with the gear retracted.

The gear indicator lights and warning horn can be tested by positioning the rotary TEST switch to LDG GEAR.

Aural Warning

A landing gear aural warning is announced if one or more of the following conditions exist:

- Gear not down and locked, both throttles retarded below approximately 70% N_2 and flaps greater than 15 degrees.
- Gear not down and locked, both throttles retarded below approximately 70% N_2 and valid radio altimeter signal indicates less than 500 feet AGL.

- Gear not down and locked, both throttles retarded below approximately 70% N_2, a non-valid radio altimeter signal exists and airspeed below 150 KIAS.

The audible warning system cannot be silenced until the situation that caused the horn to sound is rectified (no horn silence button).

NORMAL OPERATION

General

In addition to energizing the gear control valve, LDG GEAR handle movement to the UP or DOWN position, also closes the hydraulic system control valve, creating pressure as indicated by illumination of the **HYD PRESS** annunciator.

At completion of either cycle, the hydraulic system control valve opens and the **HYD PRESS** annunciator extinguishes.

Retraction

Placing the LDG GEAR handle UP, energizes the gear control solenoid valve to the retract position. The control valve is positioned to direct pressure to the retract side of each gear actuator. The down lock mechanism in each actuator releases and retraction begins (Figure14-5).

As each gear approaches the fully retracted position, it is snatched by the hydraulically loaded uplock hook that drives overcenter and assisted by spring tension. An uplock micro switch is actuated as each gear is locked in position.

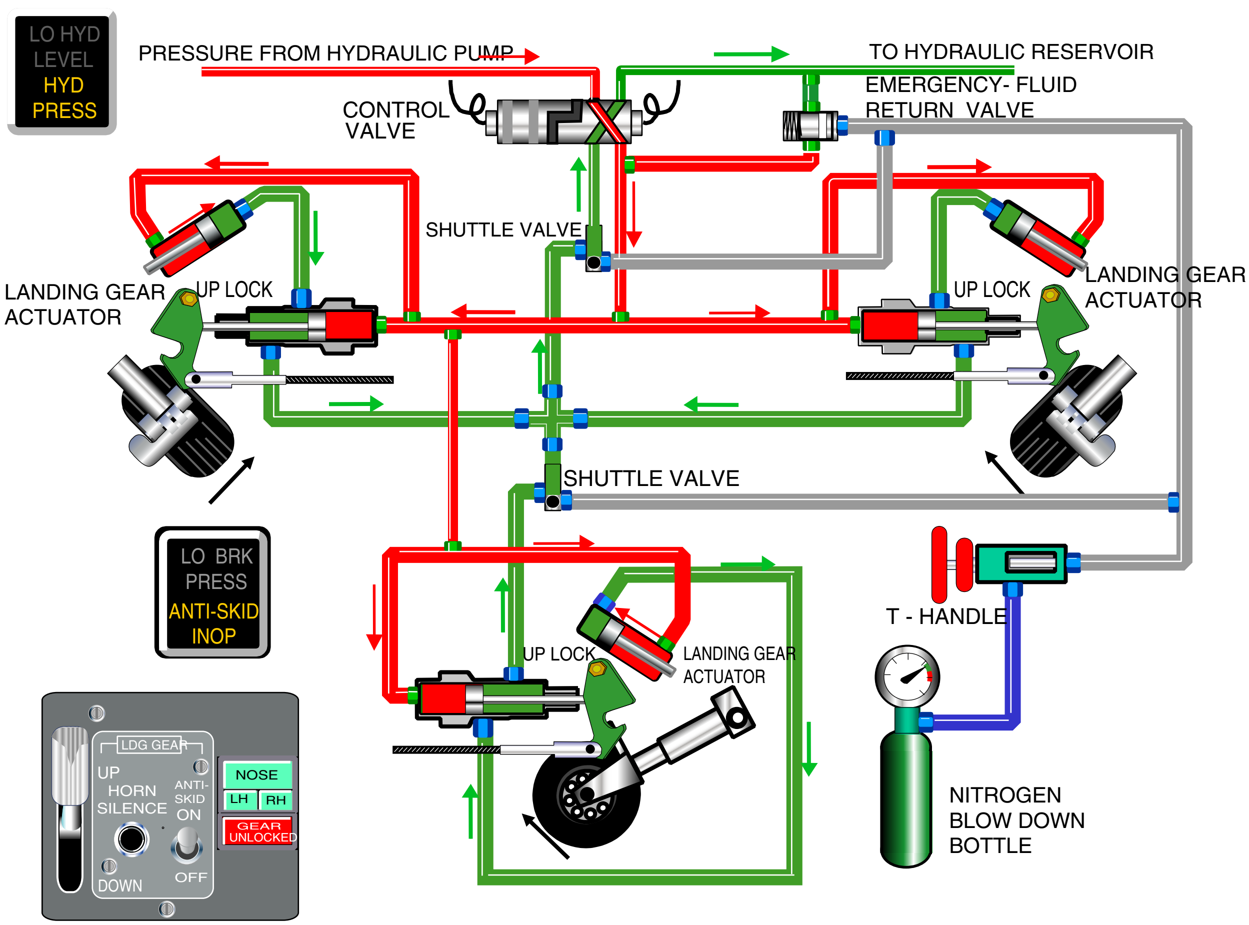


Figure 14-5 Landing Gear System, Retraction

When all three uplock micro switches have actuated (in series), the gear control valve circuit is interrupted and the valve returns to neutral bleeding off hydraulic pressure to return (Figure 14-5). The HYD PRESS annunciator should extinguish at this time and all position indicator lights on the control panel are extinguished.

Extension

Placing the LDG GEAR handle DOWN, energizes the gear control solenoid valve to the extend position. The valve is positioned to apply pressure to the uplock actuators, releasing the gear uplocks. When the uplocks release, pressure continues to the gear actuators. As each gear reaches the fully extended position, down locks within each actuator engage and down lock micro switches are actuated. When all three down lock micro switches are actuated, the control valve circuit is interrupted and the gear control valve returns to the neutral position allowing hydraulic fluid to return to the reservoir. As each gear is locked down, the respective green NOSE, LH, and RH position indicators on the gear control panel illuminate. The red UNLOCK light extinguishes when all three green indicators illuminate (Figure 14-6).

NOTE

The landing gear are locked up and down mechanically. Main system hydraulic pressure is released as all three gear are locked in position.

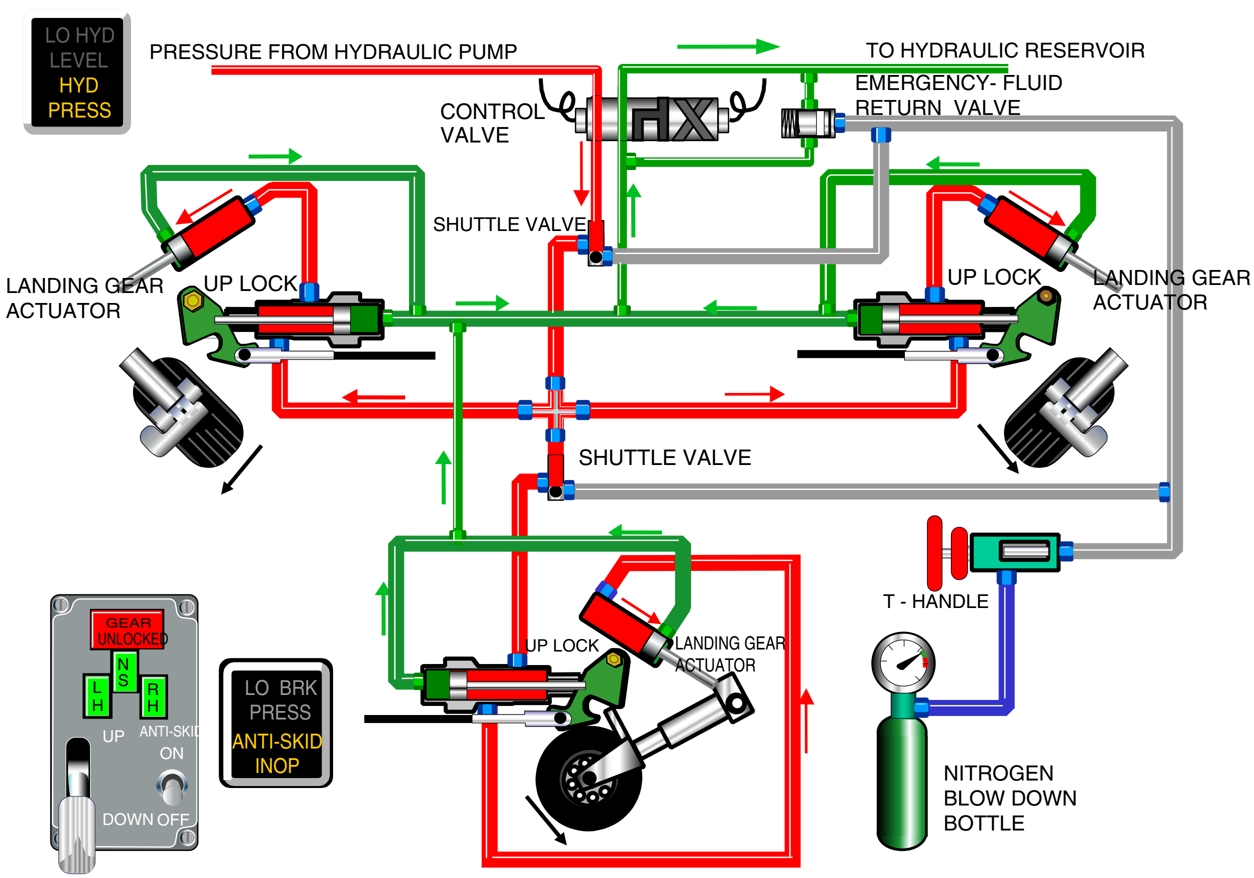


Figure 14-6 Landing Gear System, Extension

EMERGENCY EXTENSION

If the hydraulic system fails or an electrical malfunction exists that prevents the landing gear from extending, the gear uplocks can be manually released for the gear to free fall.

Emergency extension is initiated by pulling the AUX GEAR CONTROL T-handle located below the pilots instrument panel adjacent to the tilt panel (Figure 14-3).

NOTE

The AUX GEAR CONTROL T-handle is located below the pilot's instrument panel regardless of the location of the landing gear control panel.

The T-handle is connected by cables to all three uplocks. The handle is pulled out a few inches until resistance is encountered and rotated clockwise 45° to lock it in position (Figure 14-3). Pulling the T-handle out, mechanically releases the gear uplocks, allowing the gear to free-fall. If necessary, use the rudder to yaw the airplane, first in one direction, then the other to fully extend the main gear. After the gear has extended and the T-handle is locked in position (45°), the round knob (or collar) behind the T-handle, is pulled out to release emergency air bottle pressure to the gear uplocks and actuators, and at the same time, pushes a dump valve open to assure a path of fluid return to the reservoir (Figure 14-7).

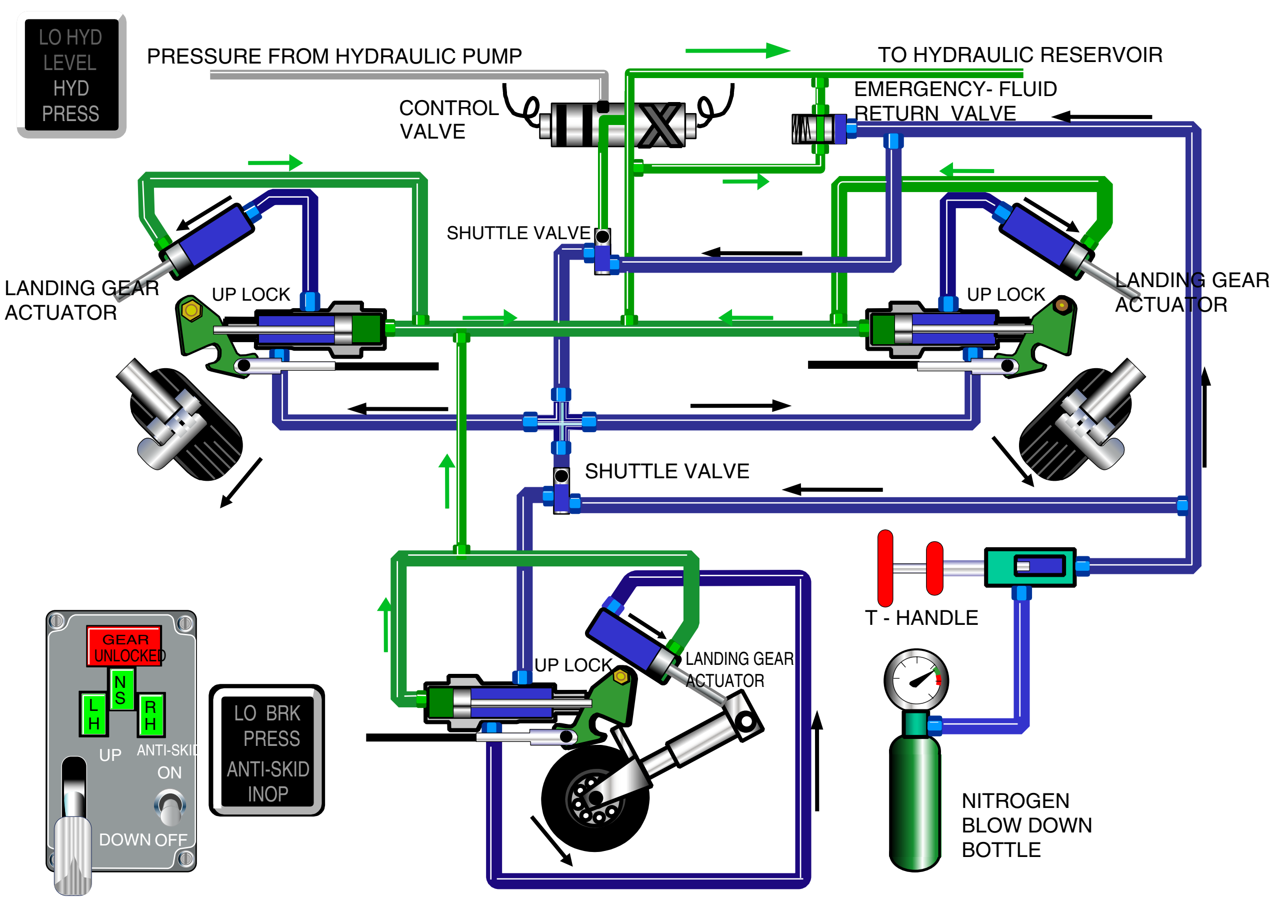


Figure 14-7 Emergency Gear Extension

The emergency air bottle is charged with nitrogen to 1,800 to 2,050 psi (Figure 14-8). It is located in left nose avionics storage compartment and is used for positive gear uplock release, gear down-locking and emergency wheel braking (Figure 14-8).

Figure 14-8 Emergency Air Bottle

Nitrogen air pressure drives the gear actuators to the full extended position where they are held by internal locks in each actuator (Figure 14-7). Once the air bottle has been actuated, hydraulic operation of the gear is not possible. Maintenance action is required after an emergency extension to restore normal operation of the landing gear.

Refer to ABNORMAL PROCEDURES, section in the FSI PTM Volume 1.

NOTE

If a gear(s) uplock fails to release mechanically after pulling the T-handle, pulling the round knob (collar) behind the T-handle will provide emergency air bottle pressure to release the uplock(s).

NOTE

If the landing gear is extended manually with the T-handle, and the three green indicator lights illuminate, emergency air bottle pressure is still required to assure positive locking of all three gear actuators.

NOSE WHEEL STEERING

Nose wheel steering is accomplished by cables connected to the rudder pedals. Nose wheel steering turning is limited by rudder stops. The turning limit is approximately 20° either side of center. A spring-loaded bungee provides additional steering capability with application of differential engine power or braking.

The nose wheel is mechanically centered during retraction. The centerline of the steering universal joint is in alignment with the centerline of the trunion supporting bolts. When the nose wheel is retracted, the lower half of the steering universal joint remains in position while the upper half, pivoting with the strut, is moved to the center position automatically centering the nose wheel. With the nose wheel fully retracted, the upper half of the steering universal joint and the nose wheel remain stationary while the lower half of the steering universal joint can move freely permitting normal operation of the rudder pedals.

During towing operations, care must be taken to ensure the nose wheel strut steering mechanism is not damaged. If the control lock is engaged (Figure 14-9) and the nose strut is deflected beyond 60º either side of center, nose steering components may be damaged. Nose strut deflection beyond 95º either side of center while towing with the control lock released may also damage the steering assembly.

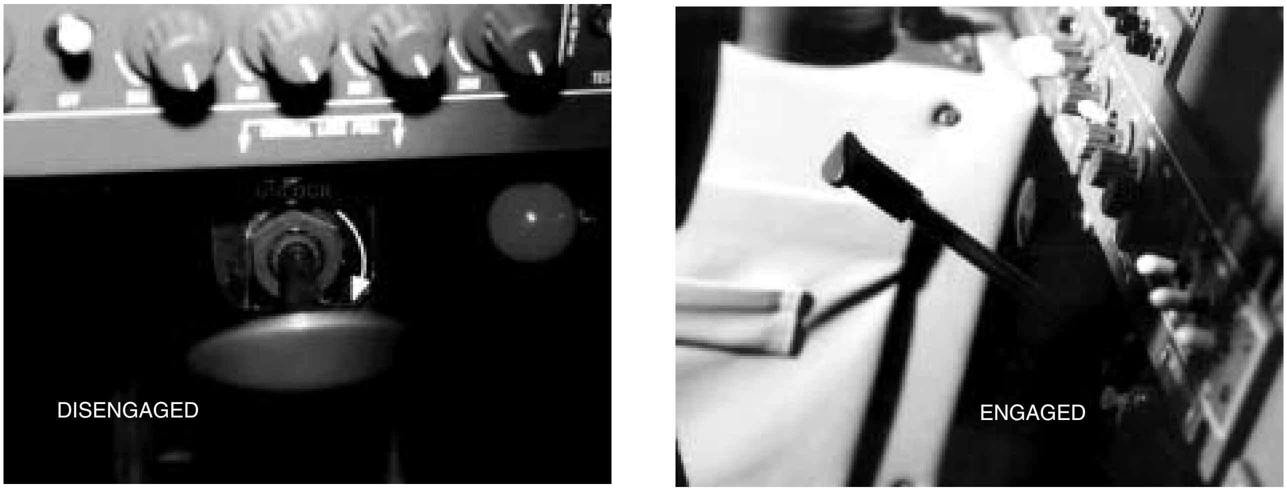


Figure 14-9 Flight Control Lock Handle

CAUTION

If the nose wheel steering bolts are sheared (indicated by loss of nose wheel steering with the rudder pedals), flight should not be attempted. the possibility exists that the nose wheel will not center after takeoff even with the gear extended.

Operating the airplane with inoperative nose wheel steering may result in violent nose wheel shimmy during takeoff and landing.

Since the nose wheel deflects with rudder pedal movement anytime the gear is extended, the pedals should be centered just prior to nose wheel touchdown during a crosswind landing.

WHEELS AND BRAKES

GENERAL

The main landing gear wheels are equipped with multiple disc carbon brakes. The brakes are actuated hydraulically by a power brake valve that is connected to each rudder pedal. The brake pedals are connected in series to permit either pilot or copilot to control the brakes.

Main wheel braking with antiskid control, is used to prevent wheel skidding on wet, dry or icy runways after a minimum wheel spinup is attained.

A parking brake valve, attached to a "pull" knob located on the pilot's lower instrument panel (Figure 14-10), is incorporated in the brake system which, when manually operated, prevents the return of hydraulic fluid pressure after the brakes are applied (Figure 14-11).

An emergency (pneumatic) braking system is provided for use when hydraulic braking fails. Auxiliary braking is controlled with a hand-operated handle (Figure 14-12) that directs equal nitrogen pressure to each brake during emergency braking conditions. High-pressure nitrogen is supplied from the emergency gear and brake pneumatic storage bottle (Figure 14-8).

WHEELS

Main Gear Wheels

Each main wheel consists of two halves to facilitate tire installation and removal. A valve assembly is installed in each outboard wheel half to inflate and deflate the tires. Fusible plugs installed in the inboard wheel halves will melt and release tire pressure if the wheel

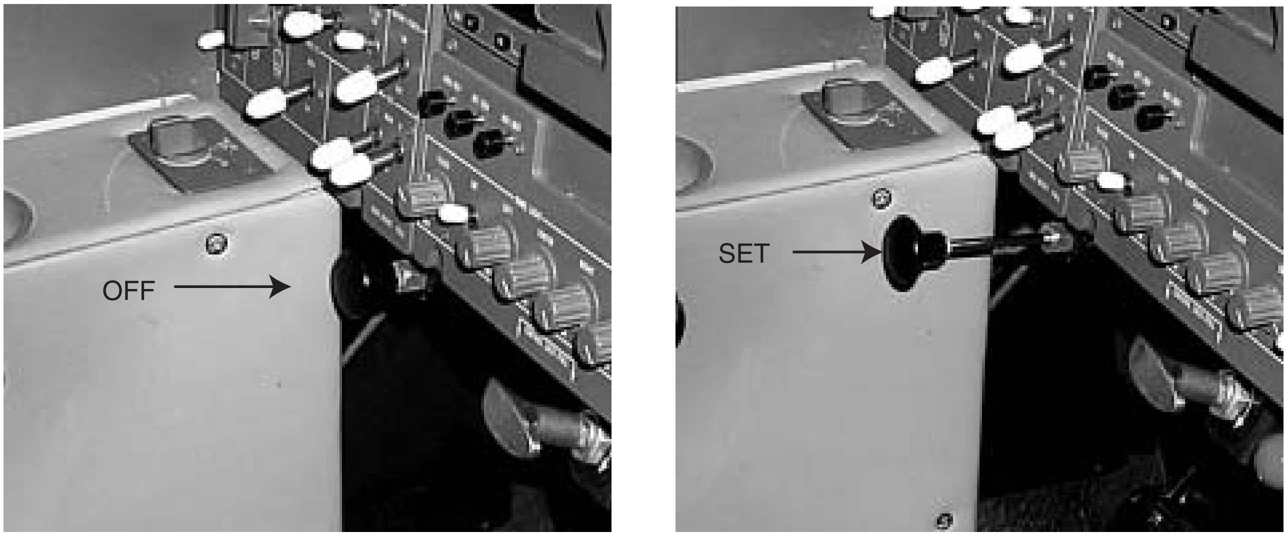


Figure 14-10 Parking Brake Knob

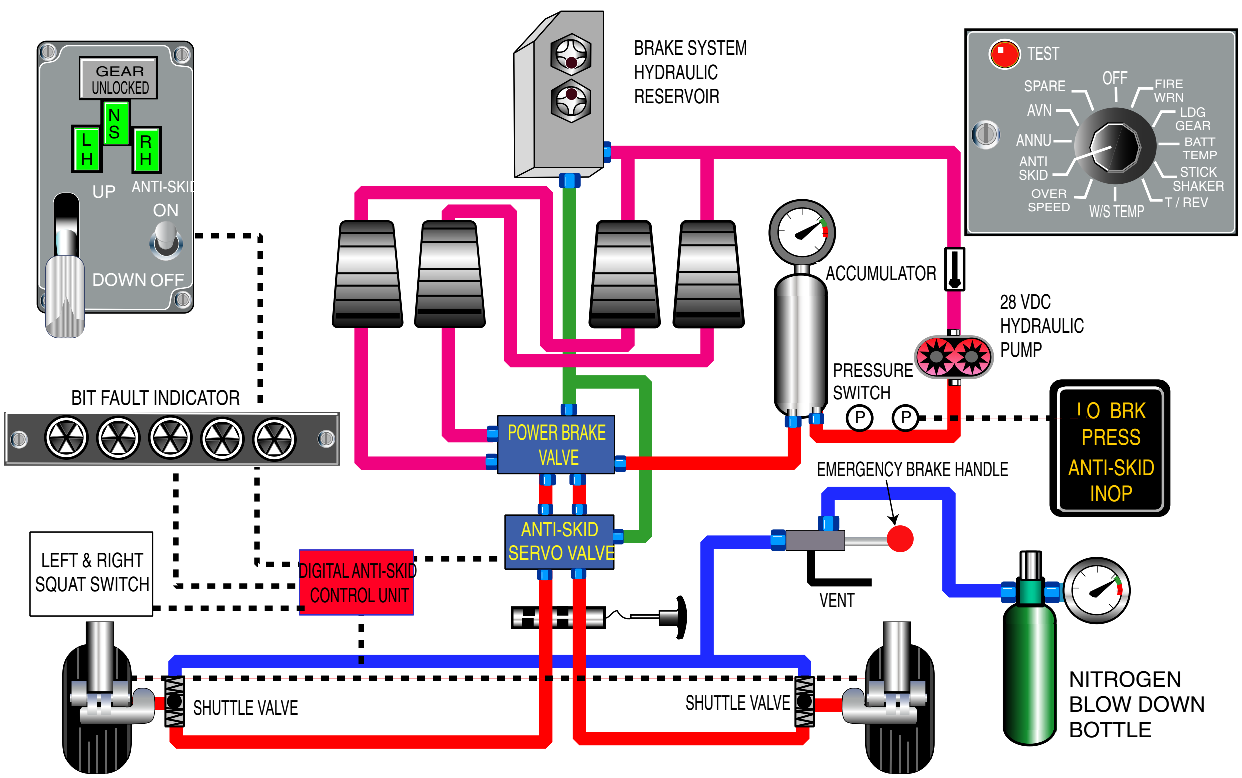


Figure 14-11 Antiskid Braking System

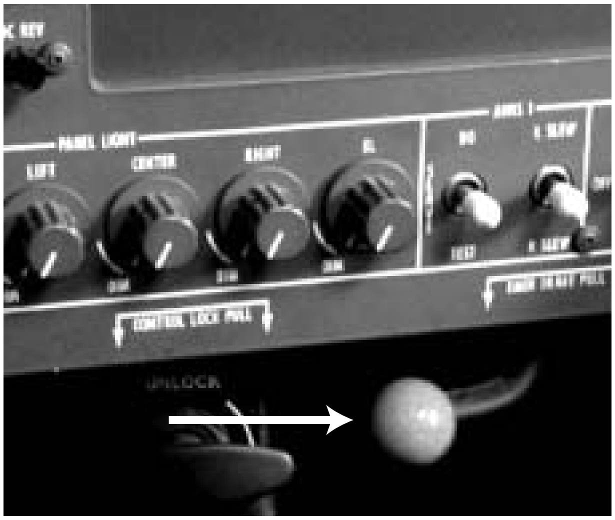


Figure 14-12 Emergency Brake Handle

is overheated to prevent the wheels from exploding. The main wheels utilize 23.5 x 8.0R12 tubeless, 14-ply rating aircraft tires, and should be inflated to 210 ±5 psig.

Nose Gear Wheel

The nose wheel assembly also consists of two halves to facilitate tire installation and removal. Each wheel half may be assembled in any position relative to one another and allows changing wheel halves without the need for rebalancing the wheel. One wheel half has a tire inflation valve. The wheel utilizes an 18 x 4.4 tubeless 10-ply rating aircraft tire, and inflation pressure should be 130 ± 5 psig.

BRAKES

General

The wheel brake system is completely separate from the main hydraulic system. It consists of an electrical control box, power brake/servo valve, motor/pump assembly, hydraulic accumulator, pressure switches, brake reservoir (Figure 14-13), mode switch, circuit breakers, indicator lights and a self test system.

The hydraulic reservoir is pressurized by cabin air pressure. The electric DC motor/pump assembly and accumulator automatically maintain constant pressure for brake operation and damp out pressure surges. The brakes are normally used as antiskid power brakes, but can be operated as power brakes without antiskid protection. In event the brake system hydraulic pressure is lost, emergency air brakes are available.

The antiskid brake system provides the crew with the option of an operative antiskid function or an inoperative anti-skid function by se-

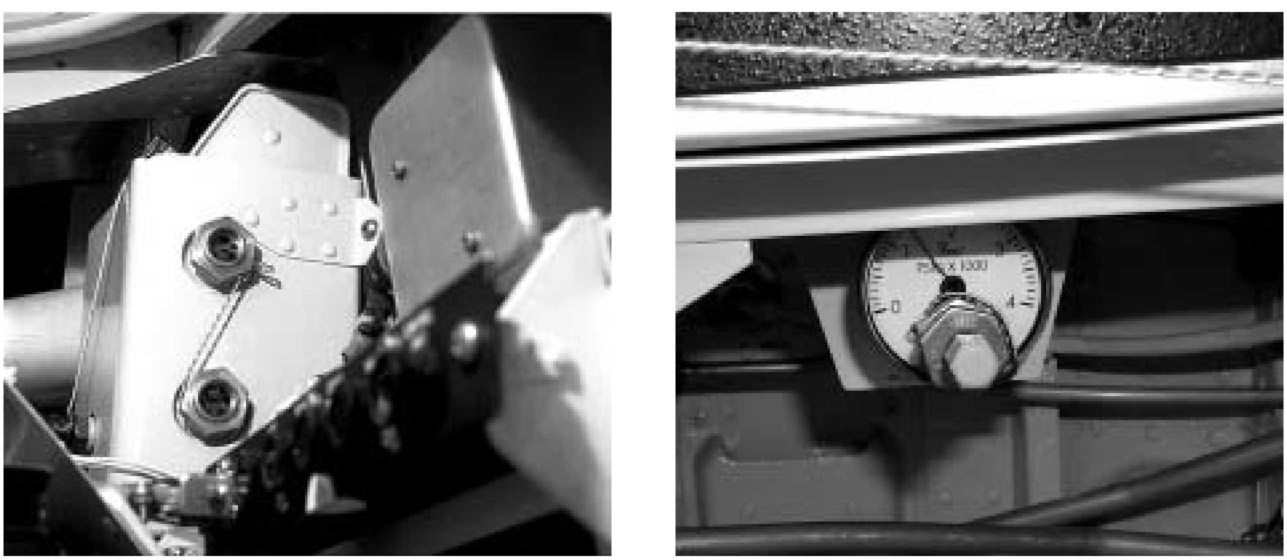


Figure 14-13 Brake Reservoir and Accumulator Gage

lecting the ANTISKID switch located on the landing gear control panel ON or OFF as appropriate (Figure 14-3).

The power brake function is operative irrespective of the antiskid control switch. The power brake function is active when the airplane's main DC buses are powered and the landing gear handle is in the extended position. In the event a main DC electrical failure occurs, causing either a total or partial antiskid/power brake failure, the brake system will not operate and function as a manual brake system. The emergency air brake system must be used.

Braking is initiated by depressing the pilot's or copilot's rudder pedal toe brakes. If both the pilot and copilot attempt to apply the brakes simultaneously, the pilot applying the greater force on the rudder pedals has control.

Digital Antiskid System

Use of the dual channel digital antiskid system permits maximum braking without wheel skid under all runway conditions. A speed transducer in each main gear wheel transmits wheel speed signals to an electronic control box (Figure 14-11). Detection of sudden deceleration of a wheel (impending skid) causes the control box to command the antiskid valve to reduce pressure being applied to the brakes individually. When the transducer signal returns to normal, braking pressure is restored to the brake(s).

A touchdown protection feature prevents touching down with locked brakes. The wheels must be rotating (some speed transducer voltage) and weight-on-wheels (both squat switches) for normal operation of the power brake and antiskid system.

A metering valve requires increased pedal force before metered pressure develops for smooth braking. Optimum braking is obtained by deployment of speed brakes at touchdown, then firmly applying and holding the brakes until the desired speed has been reached. Do not pump the brakes. The digital antiskid system continually monitors for faults and illuminates the **ANTISKID INOP** light if a fault occurs.

NOTE

> The antiskid system is inoperative with the parking brake set or using emergency braking.

Normal Operation

With the LDG GEAR handle DOWN and normal DC power available, a pressure switch controls the DC motor-driven hydraulic pump to maintain 900-1300 psi for brake operation. An accumulator dampens pressure surges. The power brakes and antiskid system receive main DC power through the PWR BRKS and SKID CONTRL circuit breakers on the left circuit-breaker panel. The hydraulic pump will periodically cycle on and off to maintain system pressure (**LO BRK PRESS** annunciator remains extinguished).

The system is supplied with fluid from the brake reservoir. Depressing the brake pedals applies pressure to actuate the power brake valve, which meters pump pressure to the brake assemblies in direct proportion to pedal force (Figure 14-11).

With the ANTISKID switch near the landing gear handle ON, the airplane on the ground, and a ground speed of at least 10-12 knots, maximum braking without wheel skid is available. Any tendency of a wheel to rapidly decelerate (skid) is detected by the digital control unit from the wheel speed transducers, and the antiskid valve is signaled to modulate braking pressure to prevent wheel skid. As wheel speed returns to normal, dumping ceases and pressure is once again increased in the brake assemblies. When the wheel speed drops below approximately 10 knots, the antiskid function disengages (**ANTISKID INOP** light remains extinguished).

Braking on each main wheel is controlled by the applicable pedal; therefore, differential braking is available.

Locked wheel protection during landing (brakes applied prior to touchdown), prevents hydraulic pressure to the brakes until both squat switches are in the ground mode for at least 5 seconds and/or the wheel speed is approximately 40 knots, whichever occurs first. Locked wheel crossover protection is also provided between the left and right wheels respectively. Ground speed greater than 40 knots, locked wheel protection will provide a pressure dump command to the slow wheel when velocity is 50 percent slower than the fast wheel.

The ANTISKID switch, located on the LDG GEAR control panel, is normally in the ON position. In the OFF position, the antiskid system is deactivated and the **ANTISKID INOP** annunciator is illuminated.

NOTE

If the ANTISKID switch is OFF, it should not be turned on while the airplane is taxiing. The antiskid system is not operative during self test and during initial power up.

If a fault develops in the antiskid system, the **ANTISKID INOP** annunciator light illuminates, and the system should be switched OFF. Brake operation remains the same except that antiskid protection is not available. When brake system pressure drops below 900 psi, the **ANTISKID INOP** and **LO BRK PRESS** annunciators both illuminate.

On the ground, testing the ANTISKID system is accomplished by momentarily selecting ANTISKID with the rotary TEST switch. The **ANTISKID INOP** annunciator and the MASTER CAUTION lights illuminate and then go out in approximately six seconds, if the test is valid.

In flight, the system is automatically tested when the landing gear is extended. Results should be the same as those obtained during the on-ground test except the MASTER CAUTION will remain extinguished and the **ANTISKID INOP** light illuminates for 3 seconds. If the antiskid system fails the self-test, the **ANTISKID INOP** light will remain illuminated.

The electronic antiskid control module incorporates test circuitry, which continually monitors the antiskid system even in flight with gear retracted. If a fault is detected, the **ANTISKID INOP** light will illuminate. Certain faults in the system are displayed on the built-in test equipment (**BITE**) indicator (fault display unit), located in the left nose avionics bay (Figure 14-14). Five circular magnetic indicators will remain in view once activated and do not change if DC power is turned off.

Figure 14-14 Antiskid Control Module, BITE Indicators

Parking Brakes

The brakes can be set by applying the brakes in the normal manner, then pulling out the PARK BRAKE handle on the left lower side of the pilot's instrument panel. This mechanically actuates the parking brake valve, trapping fluid in the brakes (Figure 14-10). Release the brakes by pushing in the PARK BRAKE handle.

NOTE

Do not set the brakes subsequent to a hard stop. Brake heat transfer to the wheel could melt the fusible plugs, deflating the tire(s).

EMERGENCY BRAKES

In event the hydraulic brake system fails, an emergency pneumatic brake system is available. The system uses air pressure from the pneumatic bottle located in the left nose equipment area. The bottle is also used for emergency landing gear extension. Air bottle pressure is adequate for stopping the airplane, even if the landing gear has been pneumatically extended.

Operation

Pulling the red EMER BRAKE PULL lever mechanically actuates the emergency brake valve (Figure 14-11). The valve meters air pressure through a restrictor and shuttle valves to the brake assemblies in direct proportion to the amount of lever movement (Figure 14-12). However, during heavy landing weights if the lever is pulled back to full extension, the restrictor should prevent a full load of high air pressure from locking the brakes.

Since air pressure is applied to both brakes simultaneously, differential braking is not possible. Returning the lever to its original position releases pressure from the brakes and vents it overboard.

NOTE

Do not depress the brake pedals while applying emergency air brakes. Shuttle valve action may be disrupted, allowing air pressure to enter the hydraulic lines and rupture the brake reservoir or apply uncommanded differential braking.

The emergency brake handle should be pulled carefully with only enough pressure to obtain the desired rate of deceleration, then held until the airplane stops. Repeated applications waste air pressure. Antiskid protection is not available during emergency braking. Do not attempt to taxi after using the emergency brakes. A fully charged air bottle is sufficient to apply approximately 10 full brake applications.

NOTE

Maintenance action is required subsequent to emergency braking.

ABNORMAL PROCEDURES - BRAKES

Wheel Brake Failure

If wheel brakes should fail, remove feet from the brake pedals and pull the EMER BRAKE handle as required to obtain the deceleration rate required.

CAUTION

ANTISKID SYSTEM DOES NOT FUNCTION DURING EMERGENCY BRAKING. EXCESSIVE PRESSURE ON THE EMERGENCY BRAKE HANDLE CAN CAUSE BOTH WHEEL BRAKES TO LOCK, RESULTING IN BLOWOUT OF BOTH TIRES.

NOTE

Use nose wheel steering for directional control.

Multiply the charted landing distance by 1.4.

Power Brake System Failure (LO BRK PRESS and ANTISKID INOP CAUTION Light on)

If power brakes fail, both the **LO BRK PRESS** and the **ANTISKID INOP** annunciators will illuminate. The PWR BRKS circuit breaker on the pilot's CB panel should be checked IN. If the annunciators remain illuminated, use the emergency brake system for landing. Same procedures, caution and notes apply as for the previous procedure, **WHEEL BRAKE FAILURE.**

Multiply the charted landing distance by 1.4

Antiskid System Failure (ANTISKID INOP CAUTION Light ON and LO BRK PRESS CAUTION Light Extinguished)

An inoperative antiskid system will be annunciated by illumination of the **ANTISKID INOP** light only. Power braking is available, but, without antiskid protection. The SKID CONTROL circuit breaker on the pilot's CB panel should be checked IN. If the CB checks IN, and the **ANTISKID INOP** annunciator remains illuminated, cycle the ANTISKID switch OFF then ON to attempt a dynamic self-test of the system.

If the ANTISKID INOP Light Remains Illuminated

Place the ANTISKID switch OFF and multiply the charted landing distance by 1.6. During landing, apply maximum reverse thrust if required (not to exceed 75% of takeoff thrust). Apply wheel braking judiously.

CAUTION

DIFFERENTIAL POWER BRAKING IS AVAILABLE. HOWEVER, SINCE ANTISKID IS INOPERATIVE, EXCESSIVE PRESSURE ON THE BRAKE PEDALS MAY CAUSE WHEEL BRAKES TO LOCK, RESULTING IN TIRE BLOWOUT.

Prepare to use emergency braking if brake pressure is depleted. If the brake hydraulic pump fails after the accumulator pressure exceeds 850 PSI, the **LO BRK PRESS** annunciator may not illuminate until normal brakes are used.

Hard Brake Pedal

If while attempting to stop the aircraft during taxi, aborted takeoff or landing, and a hard pedal is experienced (unable to depress the brake pedals), remove feet from the pedals and use the emergency braking system. This condition may be caused by a loss of DC power to power brakes or a malfunctioning antiskid system.

CAUTION

IF DURING TAXI, ABORT OR LANDING, A HARD BRAKE PEDAL - NO BRAKING CONDITION IS ENCOUNTERED, OPERATE THE EMERGENCY BRAKE SYSTEM AS REQUIRED. CORRECT PRIOR TO FLIGHT.

After using the emergency pneumatic brake system, maintenance must be performed to correct the malfunction prior to flight.

LIMITATIONS

SPEED LIMITS

Maximum Landing Gear
Extended Speed - V_{LE} ....250 KIAS

Maximum Landing Gear Operating Speed - V_{LO} (Extending) ....250 KIAS

Maximum Landing Gear
Operating Speed - V_{LO}
(Retracting) ....200 KIAS

TAKEOFF AND LANDING OPERATIONAL LIMITS

Michelin part number 031-613-8 nose tire and OM13701 main tire are the only tires approved. The nose tire must be inflated to 130 plus or minus 5 psi with weight on the wheels.

Maximum Tire Ground
Speed ....165 Knots

Takeoff and landings are limited to paved runway surfaces

Antiskid must be operational for takeoff.

HYDRAULIC FLUID

Brakes, struts and shimmy damper systems are serviced with the following fluids:

Use Skydrol 500A, B, B-4, C, or LD-4; or Hyjet, Hyjet W, III,IV,IVA, 0r IVA Plus only.

QUESTIONS

1. On the ground, the LDG GEAR handle is prevented from movement to the UP position by:
   A. Mechanical detents.
   B. A spring-loaded locking solenoid.
   C. Hydraulic pressure.
   D. A manually applied handle locking device.

2. The landing gear uplock mechanisms are:
   A. Mechanically held engaged.
   B. Hydraulically disengaged normally; or pneumatically released in an emergency.
   C. Electrically engaged and disengaged.
   D. A and B.

3. Landing gear down locks are disengaged:
   A. When hydraulic pressure is applied to the retract side of the gear actuators.
   B. By action of the gear squat switches.
   C. By removing the external down lock pins.
   D. By mechanical linkage as the gear begins to retract.

4. Each main gear wheel incorporates a fusible plug that:
   A. Blows out if the tire is over-serviced with air.
   B. Melts, deflating the tire if an overheated brake creates excessive tire pressure.
   C. Is thrown out by centrifugal force if maximum wheel speed is exceeded.
   D. None of the above.

5. At retraction, if the nose gear does not lock in the UP position, the gear panel light indication will be:
   A. Red light on, green LH and RH lights on.
   B. Red light out, green LH and RH lights on.
   C. Red light on, all three green lights out.
   D. All four lights out.

6. The gear warning horn sounds when one or more gear are not down and locked and:
   A. Flaps are extended beyond the 15° position - both throttles retarded below 70% N_2.
   B. Airspeed is less than 150 KIAS.
   C. Either throttle is retarded below 70 % N_2 rpm.
   D. Both throttles are retarded below 70% N_2 rpm and airspeed is >150 KIAS.

7. When the LDG GEAR handle is positioned either UP or DOWN:
   A. The hydraulic system control valve is energized open.
   B. The hydraulic system control valve is energized closed.
   C. The hydraulic system control valve is not affected.
   D. The **HYD PRESS** annunciator light remains out.

8. Emergency extension of the landing gear is accomplished by actuation of:
   A. A switch for uplock release and application of air pressure.
   B. One manual control to release the uplocks and apply air pressure for extension.
   C. Two manual controls - one to mechanically release the uplocks, the other to apply air pressure for gear extension and down-locking.
   D. None of the above.

9. Nose wheel steering is operative:
   A. Only on the ground.
   B. With the gear extended or retracted.
   C. With the gear extended, in flight or on the ground.
   D. None of the above.

10. The power brake valve is actuated:
    A. Mechanically.
    B. Mechanically by the emergency airbrake control lever.
    C. Hydraulically by brake pedal pressure.
    D. Automatically at touchdown.

11. Do not depress the brake pedals while simultaneously using the emergency brake system because:
    A. Manual braking will override the air brakes.
    B. The shuttle valve may allow air pressure into the brake reservoir, rupturing it or causing uncommanded differential braking.
    C. The shuttle valve will move to the neutral position and no braking action will occur.
    D. The brakes will be "spongy."

12. The DC motor-driven hydraulic pump in the brake system operates:
    A. During the entire time the LDG GEAR handle is in the DOWN position.
    B. As needed with the LDG GEAR handle DOWN in order to maintain system pressure.
    C. Only when the **LO BRK PRESS** annunciator illuminates.
    D. Even when the LDG GEAR handle is UP to keep air out of the system as the airplane climbs to altitude.

CHAPTER 15
FLIGHT CONTROLS

CONTENTS

ILLUSTRATIONS

CHAPTER 15
FLIGHT CONTROLS

INTRODUCTION

The Citation EXCEL's primary flight controls consist of manually operated ailerons, rudder, and elevators. The elevator is hinged to a two position, horizontal stabilizer that is electrically controlled and hydraulically actuated during flap selections. Primary flight controls are manually actuated by dual interconnected rudder pedals and dual interconnected conventional control columns, and can be immobilized by control locks when on the ground. Trim is mechanical in all three axes. Electrical elevator trim is also provided.

Secondary flight controls consist of dual segmented flaps on the trailing edge of each wing, a two-position horizontal stabilizer, and speed brakes on the upper and lower surfaces of each wing, all electrically controlled and actuated by the main hydraulic system. The angle-of-attack system warns of impending stalls by shaking the control columns and providing visual indication of angle-of-attack. Yaw damping is provided as a function of the autopilot.

PRIMARY FLIGHT CONTROLS

AILERONS

The ailerons provide lateral control of the airplane and are operated mechanically by control wheel movement. A trim tab wheel mechanically operates a trim tab attached to the trailing edge of the left aileron, which provides aerodynamic movement of the ailerons.

A mechanical interconnect "bungee" between the ailerons and the rudder provide small rudder deflections with aileron movement and small aileron deflection with rudder movement to enhance lateral stability. The bungee can be manually overridden by cross controlling.

RUDDER

The rudder provides control of the airplane about the vertical axis and is controlled mechanically by dual rudder pedals. The trim tab on the rudder trailing edge is controlled mechanically by a rudder trim wheel located on the rear portion of the center pedestal.

The rudder pedals operate the rudder, nose wheel steering and brakes. Pushing on the lower part of the pedal operates the rudder and steering; pushing on the upper part operates the brakes. The pilot's and copilot's pedals are interconnected through transfer tube assemblies. Movement from one position transfers movement to the other position. Each rudder pedal may be adjusted to three different positions by pushing an adjustment lever mounted on the inboard side of each rudder pedal and moving the pedal to the desired position.

RUDDER BIAS

AFM Configuration Codes	Effectivity by Serial Number
AB	Airplanes equipped with Rudder Bias
AC	Airplanes not equipped with Rudder Bias

General

The rudder bias system was developed to increase rudder travel from 22 degrees to 28.5 degrees either direction without requiring excessive pressure on the rudder pedals. The advantage of rudder bias is that it significantly decreases VMCG speeds which results in lower V1 speeds. The overall benefit is shorter takeoff field length requirements, especially during wet runway conditions.

The rudder bias system is comprised of separate left and right pneumatic lines plumbed into one dual actuating cylinder.

A closed loop cable system is connected to the Rudder Bias Actuator at one end and to the rudder sector at the opposite end (Figure 15-1). The cable is driven by the rudder bias actuator to rotate the rudder sector left and right. The rudder sector drives the rudder directly which also indirectly drives the primary cable loops and the autopilot servo. With approximate equal thrust set on both engines, pneumatics are balanced and do not affect rudder position when acting equally together.

The rudder bias system is designed to automatically engage upon loss of one engine thrust. The pneumatic actuator, powered by engine bleed air, will pull the rudder into position and compensate for asymmetric thrust (Figure 15-2). The rudder bias system delivers rudder assist for the pilot to compensate for adverse yaw during unbalanced engine thrust conditions, especially if an engine failure occurs during takeoff.

Operation

The rudder bias system is automatically powered ON by main DC electrical through the RUDDER BIAS circuit breaker on the pilot's circuit breaker panel. A **RUDDER BIAS** caution light will illuminate to indicate system malfunctions.

The Bias Actuator Shutoff valve, (Figure 15-1), opens electrically and bleed air from the engines flow through the shutoff valve to port left and right engine bleed air to the left and right

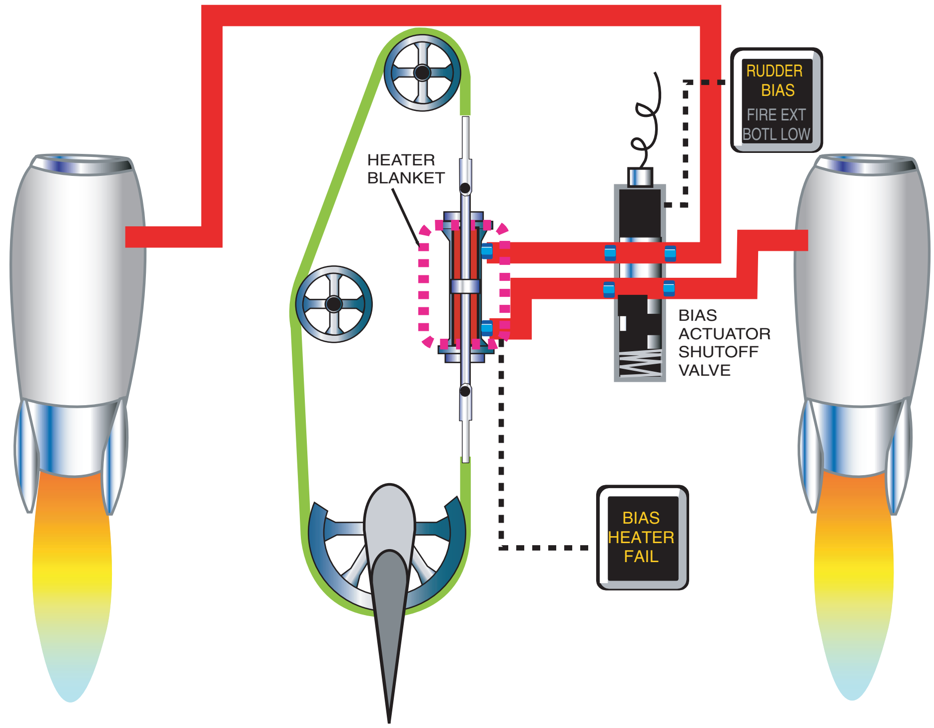


Figure 15-1 Rudder Bias System, Both Engines Equal Power

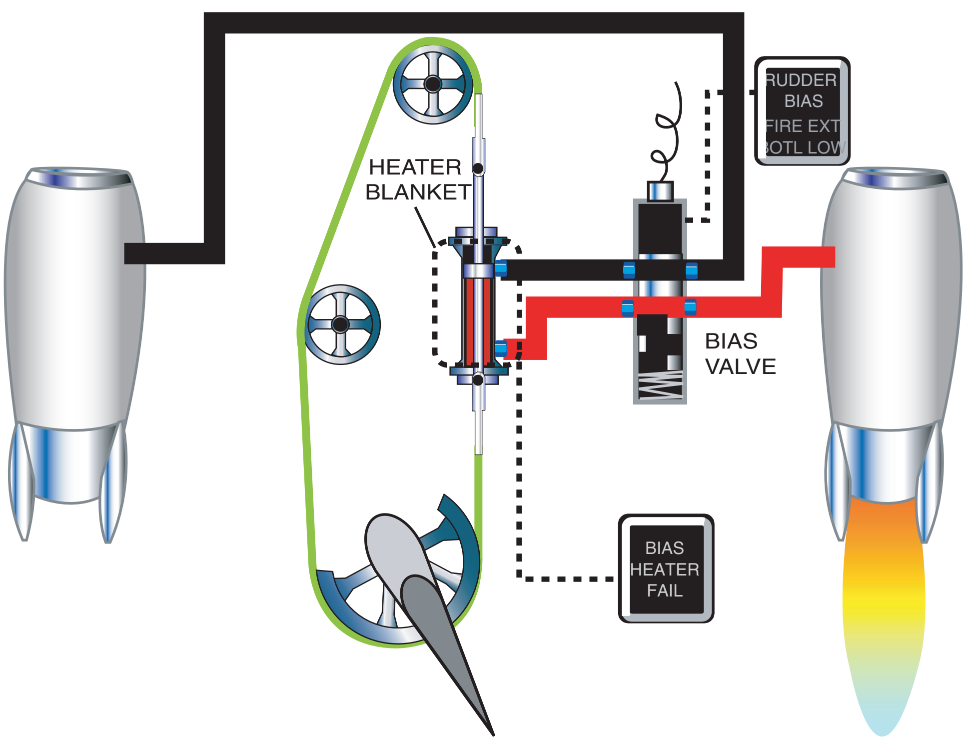


Figure 15-2 Rudder Bias System, One engine Shutdown

command halves of the rudder bias actuator. The rudder should be in a neutral position during periods of equal thrust on both engines.

If thrust is unequal, the rudder bias actuator will automatically drive the closed loop cable system to move the rudder toward the engine developing the higher thrust, thereby assisting the pilot to counteract adverse yaw (Figure 15-2).

The position of the valve is monitored by the **RUDDER BIAS** caution light and will illuminate if the valve fails. When power is removed from the valve, bleed air from both engines is shutoff and both command halves of the actuator are vented to atmosphere.

NOTE

The shutoff valve is commanded closed if either or both thrust reversers are deployed normally or emergency stowed.

Rudder Bias Heater Blanket

An electrical powered dual element Rudder Bias Heater Blanket is installed around the rudder bias actuator to prevent it from freezing. Normally, the actuator is warmed by engine bleed air flowing through the command halves and vented overboard. Electrical power for the two heating elements is controlled separately by dedicated thermostats for redundancy. If the actuator temperature drops below 40°F, power is applied to the heaters, and shuts off as temperature increases above 60°F. This cycling action may occur if engine power is reduced in extremely cold environmental conditions, such as a descent from altitude.

An amber **BIAS HEATER FAIL** light located on the center instrument panel, to the right of the Flight Director Mode Panel adjacent to the FD/AP PFD 1/2 switchlight (Figure 15-3), monitors the thermostats. The light will flash if a thermostat is defective. Depressing the light will cause the light to illuminate steady.

NOTE

Figure 15-3 BIAS HEATER FAIL Light

Illumination of the **BIAS HEATER FAIL** light will not activate the MASTER CAUTION lights.

The heater blanket is tested upon initial power up by the **BIAS HEATER FAIL** light illuminating momentarily and extinguishing. If the light fails to extinguish, dispatch is prohibited until the malfunction is corrected. If the light illuminates in flight, the flight may continue in a normal manner.

ELEVATORS

The elevators provide longitudinal control of the airplane and are operated mechanically by fore and aft movement of the control wheels. A trim tab is located on the trailing edge of each elevator and may be electrically or manually operated.

CONTROL LOCK SYSTEM

The control lock, when engaged, locks the primary flight controls in neutral and both throttles in cutoff. Prior to engaging the control lock, ensure the nose wheel is aligned fore and aft, move both throttles to CUT OFF and neutralize the flight controls. Rotating the CONTROL LOCK handle (Figure 15-4), on the

pilot's lower switch panel, 45° clockwise and pulling out until the handle returns to the horizontal position locks the flight controls in neutral and the throttles in CUT OFF.

To unlock the flight controls and throttles, rotate the handle 45° counter clockwise and push in until it returns to the horizontal position.

NOTE

The aircraft should not be towed with the control locked engaged.

Figure 15-4 Control Lock Handle

TRIM SYSTEMS

GENERAL

rudder and aileron trim are operated manually by cables from trim wheels in the cockpit. A mechanical trim wheel and electrical trim switches are provided for the elevators.

RUDDER AND AILERON TRIM

Rudder and aileron trim is initiated by rotation of the aileron trim and rudder trim wheels on the pedestal (Figure 15-5). Cable connections transmit motion to position the tabs. A mechanical indicator adjacent to each trim wheel indicates direction of trim input, nose deflection left or right, left/right wing down. The rudder tab is a servo boost tab. It provides a boost to the rudder

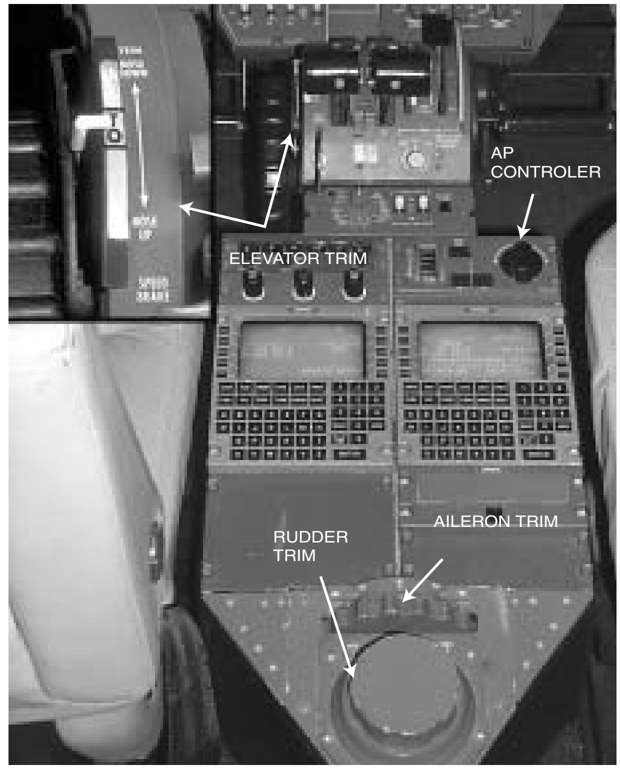


Figure 15-5 Manual Trim Wheels

when the rudder is displaced from neutral. For each degree of rudder deflection, the tab will deflect one-half that amount in the opposite direction.

ELEVATOR TRIM

Manual Trim

Manual elevator trim is initiated by rotating the elevator trim wheel (Figure 15-5). Motion is mechanically transmitted to position the elevator tabs. As the tabs move, a pointer on the elevator TRIM indicator moves toward the NOSE DOWN or NOSE UP position, as applicable.

Electrical Trim

Electrical trimming of the elevators is accomplished with a split-element pitch trim switch on the outboard side of each control wheel (Figure 15-6). Both elements of the switch must be moved simultaneously to complete a circuit to the electric trim actuator in

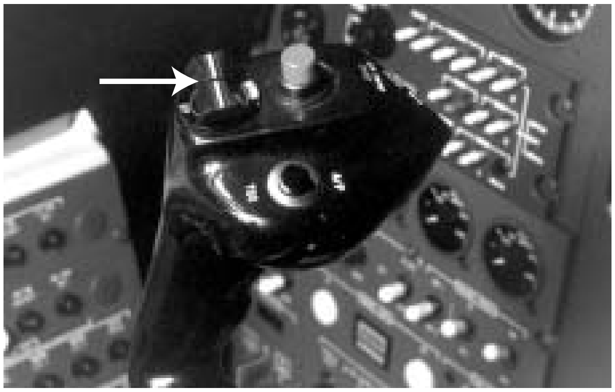


Figure 15-6 Pitch Trim Switch

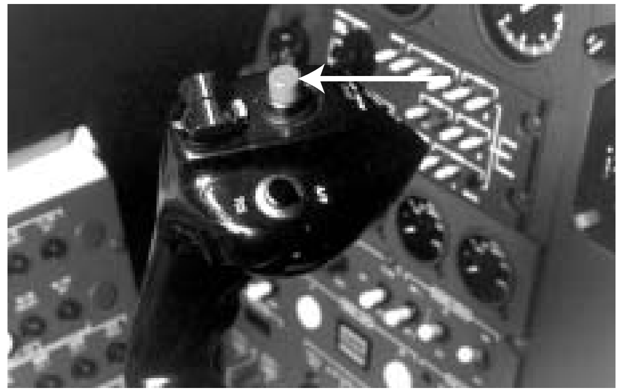


Figure 15-7 AP/TRIM DISC Switch

the tail cone. The pilot's pitch trim input overrides those made by the copilot.

As the trim switch is moved UP or DOWN, the elevator is positioned to adjust pitch attitude as indicated by the elevator TRIM indicator.

Prior to flight, the system can be checked for proper operation by moving both elements of the switch simultaneously in both directions, noting that trim occurs in the appropriate directions. Check for malfunctions by attempting to trim with one element of the switch. If trimming occurs, the system is malfunctioning and must be restored to normal operation prior to flight.

Runaway or malfunctioning trim can be interrupted by momentarily depressing the AP/TRIM DISC switch (Figure 15-7) on the control wheel(s) and pulling the PITCH TRIM circuit breaker on the left circuit breaker panel to remove electrical power from the system.

NOTE

If the airplane is on the ground and the elevator is positioned out of the take-off range (white decal), the NO TAKE-OFF annunciator will illuminate.

NOTE

Do not engage the autopilot with electric trim inoperative.

SECONDARY FLIGHT CONTROLS

GENERAL

The secondary flight controls consist of wing flaps, a two position horizontal stabilizer, and speed brakes; all are electrically controlled and hydraulically actuated (Figure 15-8).

All secondary flight controls are operated by switches and levers on the throttle pedestal.

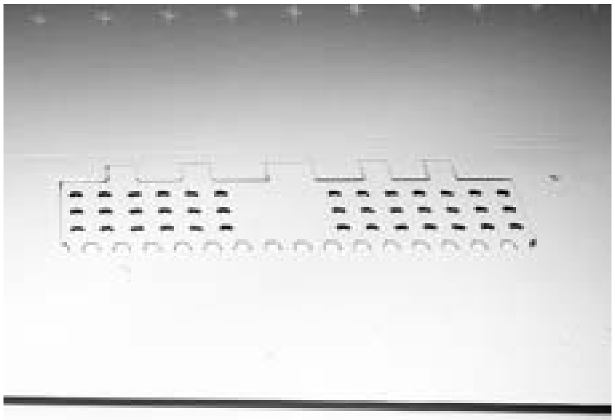
SPEED BRAKES

General

There are two speed brake segments on the top and bottom of each wing (Figure 15-8). The speed brakes provide fast, precise speed control that extend and retract between 0.0 and 1.5 seconds. The speed brakes are positioned either retracted or extended. The system includes two hydraulic actuators, a solenoid valve, four speed brake segments, a safety valve, and a white **SPD BRK EXTEND** advisory annunciator (on the ground will illuminate the NO TAKEOFF caution annunciator). The speed brakes are electrically controlled by a switch on the throttle pedestal below the throttles (Figure 15-9).

Operation

Placing the speed brake switch momentarily in the EXTEND “down” position (the switch will



SPPED BRAKE, TOP OF WING

SPEED BRAKE, BOTTOM OF WING

HORIZONTAL STABILIZER
POSITION INDICATORS

FLAPS

Figure 15-8 Secondary Flight Controls

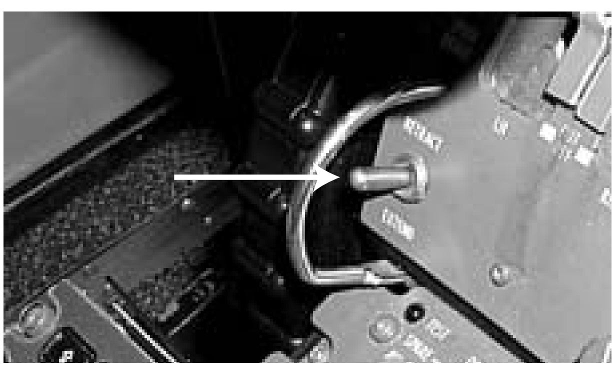


Figure 15-9 Speed Brake Switch

spring-load to center when released) will energize the hydraulic system control valve closed, providing pressure as indicated by illumination of the **HYD PRESS** annunciator. The speed brake solenoid valve is energized, directing pressure to force the speed brakes out of their mechanical retainers for extension. The safety valve, in parallel with the control valve, is energized closed (Figure 15-10).

When the speed brakes are fully extended, the white **SPD BRK EXTEND** annunciator illuminates. Simultaneously, the hydraulic system control valve opens to relieve pressure, and the amber **HYD PRESS** annunciator extinguishes. The solenoid valve returns to neutral, blocking all fluid lines to the actuators, thus hydraulically locking the speed brakes in the extended position (Figure 15-10).

The SB switch is momentarily positioned to RETRACT "up" to retract the speed brakes. The

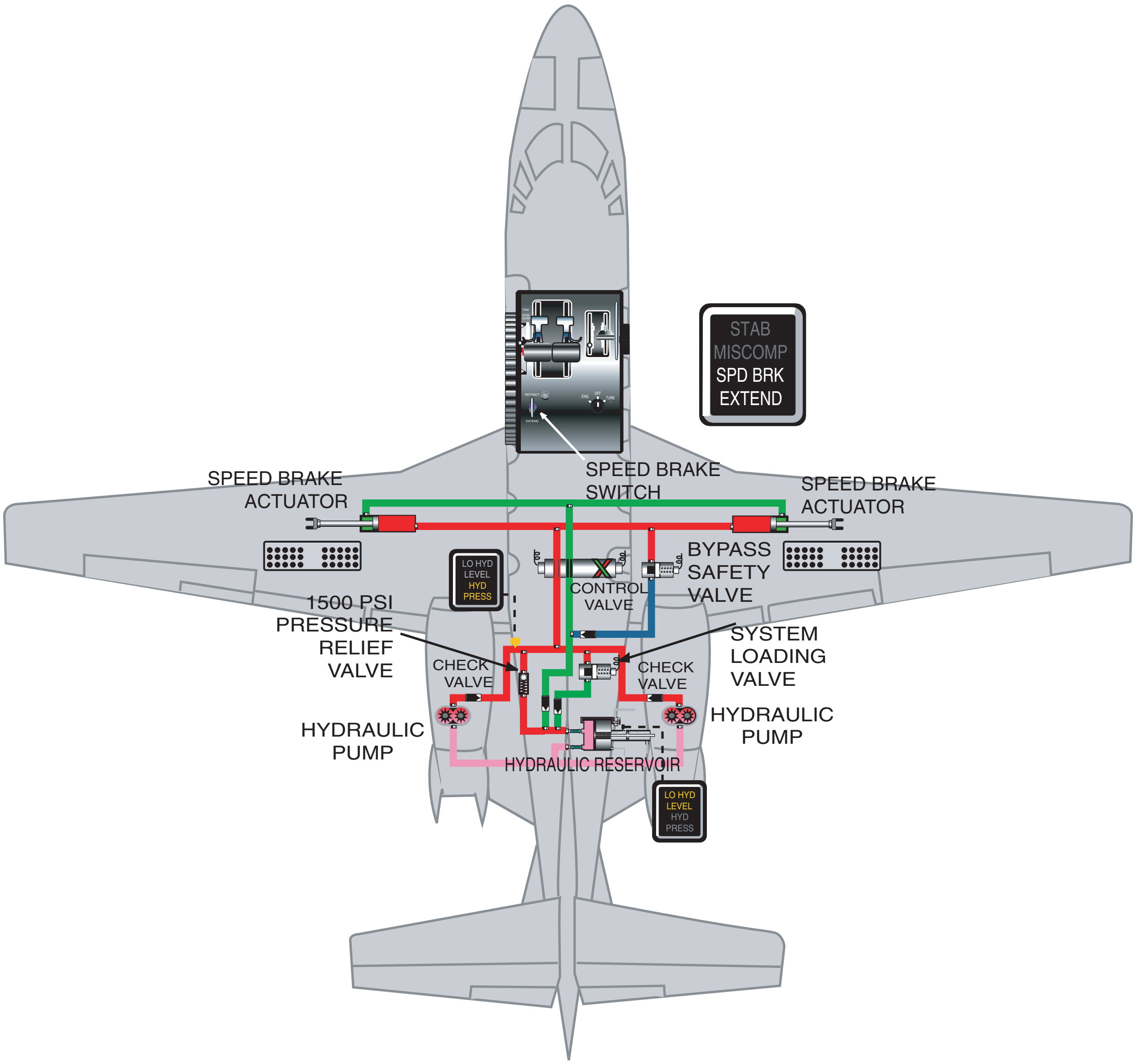


Figure 15-10 Speed Brake System Schematic

hydraulic system pressurizes, the safety valve de-energizes open (bleeds off fluid), and the speed brake solenoid valve is positioned to direct pressure for retraction. The **SPD BRK EXTEND** annunciator extinguishes. The speed brakes retract into mechanical retainers, and the hydraulic system depressurizes.

The speed brake retainers consist of two pins on each lower speed brake panel that are forced into retaining clips in the under side of the wings to prevent droop after hydraulic pressure is removed.

If either throttle is advanced past approximately 80-85% N_2 with the speed brakes extended, circuitry is completed to the solenoid valve and the main hydraulic system control valve for speed brake retraction.

If electrical failure occurs with the speed brakes extended, the safety valve opens releasing trapped fluid, allowing the speed brakes to blow to a trail position. If electrical failure occurs with the speed brakes retracted, they cannot be extended.

If main hydraulic system failure occurs while the speed brakes are extended, they will remain extended until the retract position is selected by the control switch or the throttles are advanced beyond approximately 80% N_2. Selecting RETRACT will allow the airload to blow the speed brakes to a trail position.

FLAPS

General

Two laminated, graphite composite Fowler type flaps on each wing can be positioned from 0° to 35°. The flaps increase lift of the wing when partially extended and increase drag and lift to help reduce speed when fully extended (Figure 15-8).

Mechanical interconnection between the left and right inboard wing flaps prevent asymmetrical flap positions and permit flap operation if one hydraulic actuator fails.

The flap selector lever, detented at the 7° and 15° positions, can be set to position the flaps anywhere between zero and 35°. Flap position is shown on a pointer to the left of the flap lever. Flap movement mechanically positions the indicator. The flaps are electrically controlled from emergency DC power through the FLAP CONTROL circuit breaker on the pilot's CB panel and the flap handle on the throttle pedestal.

Operation

Moving the flap lever causes the hydraulic system control valve to close increasing hydraulic pressure as indicated by illumination of the **HYD PRESS** annunciator. The flap solenoid valve energizes, directing pressure for flap operation. When the flaps reach the selected position, the hydraulic system control valve opens to relieve hydraulic pressure and the flap solenoid valve deenergizes to its neutral position. In the neutral position, the valve blocks all fluid lines to the actuators, hydraulically locking the flaps in the selected position.

In the event of electrical failure, the flap solenoid valve remains in the neutral position, and flap position cannot be changed.

If hydraulic system failure occurs with the flaps retracted, they cannot be extended. With the flaps in an extended position, the flaps will remain in the selected position unless the handle is moved. Once the solenoid valve is energized, the flaps may "blow upward" to a trail position as determined by air loads present.

Flap extension time from 0° to 35° is 16 to 20 seconds. Retraction time from 35° to 0° is 17 to 21 seconds.

The **NO TAKEOFF** annunciator will illuminate when the flaps are set less than 7° or greater than 15° and the airplane is on the ground. Flap-gear warning switches mounted in the throttle quadrant provide in-flight aural warning when all three gear are not down and locked and the flaps are lowered beyond 15° and both throttles retarded below approximately 70% N_2

TWO POSITION HORIZONTAL STABILIZER

General

The two-position horizontal stabilizer trim system consists of a movable horizontal stabilizer to improve flight characteristics. Three subsystems comprise the stabilizer trim system: (1) An electrical control system; (2) an electronic monitoring system; (3) and a self-contained hydromechanical motor, gearbox and screw assembly actuator. The stabilizer is pivoted about an aft mounted hinge and actuated by a connection at the stabilizer front spar by a hydromechanical actuator (Figure 15-11).

The stabilizer is actuated to one of two positions only, +1° (cruise) or -2° (take-off and landing) angle of incidence. Stabilizer position depends on flap handle position and airspeed. Flaps selected UP result in a stabilizer incidence of +1° (cruise position). Flaps selected in any position other than UP will result in a stabilizer incidence of -2°(landing position). Intermediate

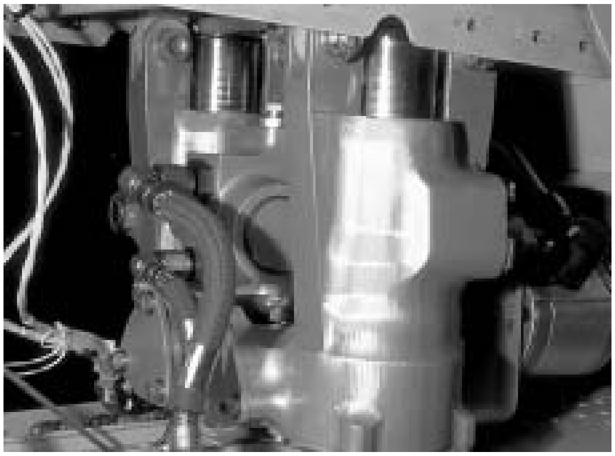


Figure 15-11 Hydromechanical Actuator

positions will result only from failures in the system. If airspeed is greater than 215 knots, ±10 knots, and, if the flight crew should inadvertently select the flap handle down, an airspeed switch disables an arming valve which prevents stabilizer movement to the -2° position.

NOTE

The crew does not have direct control of positioning the stabilizer. Stabilizer movement is only controlled by the position of the flap handle.

An annunciator panel warning light will alert the flight crew of incorrect horizontal stabilizer position. A **STAB MIS COMP** caution light will illuminate if flaps have been selected UP or DOWN and the stabilizer does not achieve the proper position within 30 seconds.

The **NO TAKEOFF** annunciator will illuminate if the aircraft is on the ground and the horizontal stabilizer is not at the takeoff/landing (-2°) position. If the NO TAKEOFF light is not recognized, advancing the throttles above approximately 85% N_2 will trigger an aural warning and “flash” the NO TAKEOFF annunciator, and cause the MASTER CAUTION RESET lights to illuminate. The takeoff warning system is independent of the stabilizer monitoring system.

Operation

The two position horizontal stabilizer is controlled electrically (emergency DC power) by a combination of switches, solenoid operated hydraulic valves, an airspeed sensor and relay.

Two independent switches (one monitoring the flap handle position and one monitoring the stabilizer position, -2°) mounted in the throttle quadrant are activated by the flap lever in any position except UP. Both switches must be in agreement to satisfy the monitor electronics or the **STAB MIS COMP** light will illuminate. Multiple structural failures or improbable failures would have to occur to prevent the switches from agreeing. In the unlikely event that a failure did occur, the fail-safe mode would result in a stabilizer command to the +1°(cruise) position. It has been demonstrated during flight test that the airplane can be landed safely with the stabilizer in the cruise position.

Stabilizer Position Limit switches located on the stabilizer rib detect the stabilizer position (+1° or -2°). When the flap handle is moved UP or DOWN, the flap switches control power, to the stabilizer position switches, and a ground will be supplied to power the hydraulic control valve closed, building up system hydraulic pressure and illuminate the **HYD PRESS** annunciator Hydraulic pressure is directed to the stabilizer solenoid control valve which is energized to provide pressure to the CRUISE port or the takeoff and landing (TO/L) port of the Horizontal Stabilizer Actuator (HSA). As the stabilizer reaches the proper position, the limit switches will cause the system to deenergize and depressurize, and the stabilizer will remain in position (Figure 15-12).

The Solenoid Control Valve is designed fail safe. If DC electrical power is removed, the valve returns to center and both cylinder ports are connected to return. If both solenoids should energize, the control valve will return to center, same as the de-energized position.

The ground circuits (energized mode) to the stabilizer control valve are open when the landing gear is in transit. The gear has priority

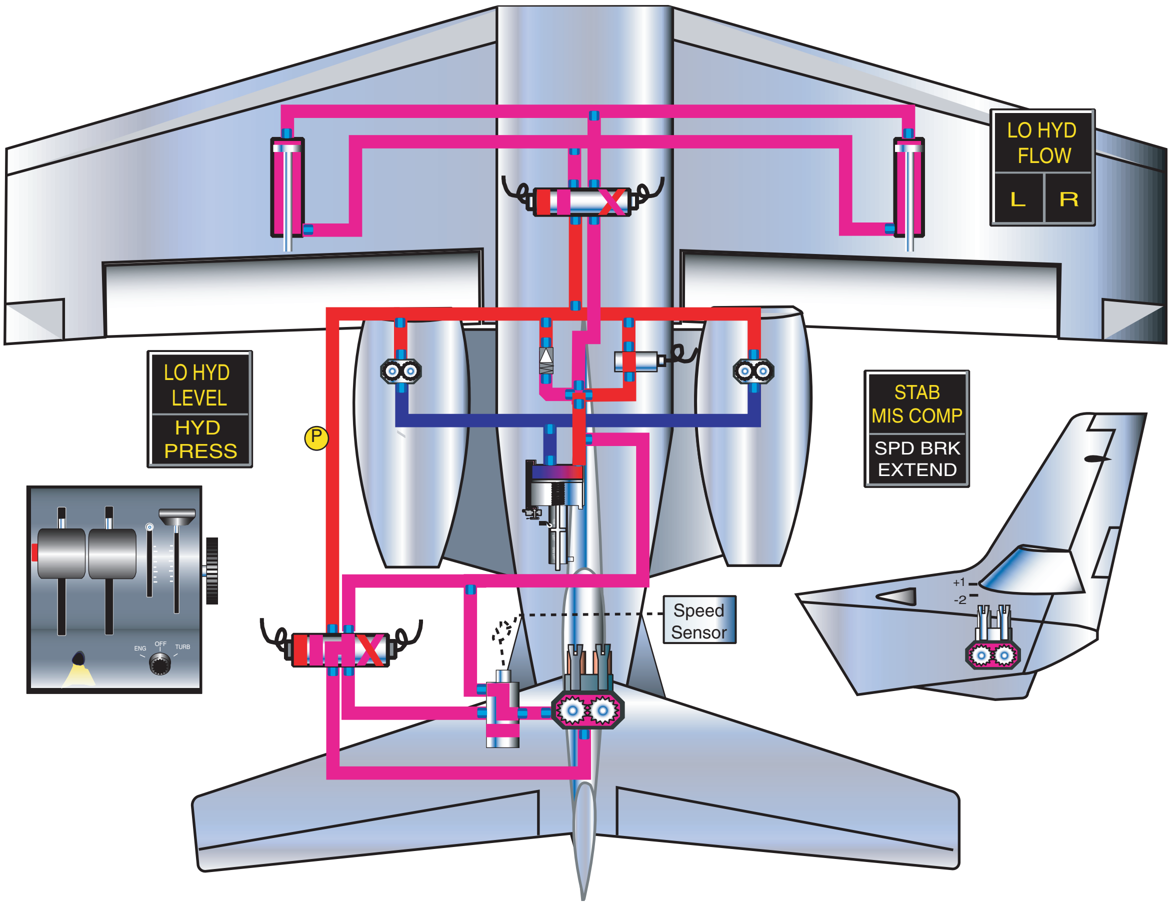


Figure 15-12 Two Position Horizontal Stabilizer Schematic

for hydraulic flow/pressure if both systems are selected simultaneously. When the landing gear is up and locked, and no longer require hydraulic pressure, the ground circuit to the solenoid valves are completed allowing the stabilizer to move when the flaps are selected UP. If the stabilizer has not achieved the proper position within 30 seconds, the **STAB MIS COMP** annunciator will illuminate.

NOTE

If the flap handle is moved UP immediately after the gear handle is raised, the flaps will move UP, but the Horizontal Stabilizer will not move until all three gear are up and locked.

The Horizontal Stabilizer Actuator (HSA) is a dual load path hydromechanical actuator that moves a self-contained screw assembly to move the horizontal stabilizer from one position to the other (+1° or -2°). The HSA is capable of withstanding ultimate loads and will not change position or creep with all power removed from the actuator motor.

A Solenoid Arming Valve protects the stabilizer from uncommanded stabilizer movement. When the valve is de-energized, the HSA remains in the CRUISE position (+1°). The arming valve must be energized to connect pressure to the TO/L port of the HSA (-2°). The arming valve is a safety feature to prevent the stabilizer from driving out of the CRUISE position at airspeeds above critical buffet speed (Figure 15-12).

The arming valve is controlled by the airspeed sensor (standby pitot-static system) and airspeed relay, and is enabled below 215 ±10 KIAS.

NOTE

In flight, with the flap handle in the UP position; if the **STAB MIS COMP** warning light illuminates, limit speed to 200 KIAS and altitude to FL410.

STALL WARNING

Aerodynamic stall warning consists of a stall strip on the leading edge of each wing (Figure 15-13), and a stick shaker operated by the angle-of-attack (AOA) system. The stall strips create turbulent airflow at high angles of attack, causing a buffet to warn of approaching stall conditions if the AOA is inoperative.

An AOA indicator on the pilot's instrument panel, actuated by signals from the angle-of-attack probe (Figure 15-14) on the right forward side of the fuselage, provides visual indication of airplane angle- of-attack. The indicator can be used as a secondary reference for approach speeds (1.3 V_{S1}) at all airplane weights and CG locations, and at all flap positions, i.e., takeoff/approach and landing. It does not replace the airspeed indicator as the primary instrument.

Two stick shaker motors, one attached to each control column, vibrates the columns as stall conditions progress. The shaker motors are energized by the angle-of-attack system. The shakers actuate at an ANGLE-OF-ATTACK indication of approximately 0.79 to 0.88, depending on flap setting and rate of deceleration.

Landing gear squat switches disable the shakers when the airplane is on the ground. Stall warning requires main DC electrical power.

The system is tested prior to flight by positioning the rotary TEST switch to STICK SHAKER. The test bypasses the squat switches

Figure 15-14 Angle of Attack Vane

Figure 15-13 Wing Leading Edge Stall Strip

and applies a high angle-of-attack signal, causing the shaker motors to operate. If the stick shakers are inoperative, dispatch is prohibited.

An approach indexer (Figure 15-15), mounted on the pilot's glareshield, provides a "heads-up" display of deviation from the approach reference. The display is in the form of three illuminated symbols which are used to indicate the airplane angle of attack. The indexer lights will display angle-of-attack deviation as follows: GREEN-on speed; RED-too slow; YELLOW-too fast.

Figure 15-15 AOA Indexer Light

AUTOPILOT SERVOS

PRIMARY CONTROL SERVOS

All primary control cables are attached to autopilot servo actuators. The pitch, roll and yaw servos are electrically driven and provide surface displacement proportional to input signals from the autopilot computer servo amplifier.

Each servo includes an engage clutch, which disengages the servo output shaft and leaves it free to rotate when the autopilot is off. The output shaft is connected through the servo drum to the airplane control cables. Electronic actuation of the primary controls is accomplished by engaging the autopilot. The electrical powered servos will transmit primary cable movement to the ailerons, rudder and elevators. This movement will also be transmitted to the rudder pedals and the control columns.

Each autopilot servo has an override function that allows the pilot to physically overpower the servo(s) by manually moving the control column or the rudder pedals, thereby disengaging the associated clutch.

ELEVATOR TRIM SERVO

Electrical elevator trim is accomplished by a trim servo. The servo is actuated electrically by the pilot or copilot control wheel switches or autopilot input and monitored by the electric trim logic module. Selecting UP or DOWN on the control wheel(s) or autopilot trim inputs will engage the electric motor on the trim servo to drive the trim tabs in the appropriate direction. Motion is also transmitted to the elevator trim wheel on the throttle pedestal.

YAW DAMPING

Yaw damping is a function of the autopilot control and the rudder servo, consisting of automatic application of rudder against transient motion in the yaw axis. With the autopilot engaged, the yaw damper is engaged automatically. The yaw damper can be engaged independent of the autopilot by depressing the YD ENGAGE switch on the autopilot control panel. An operative yaw damper is not required for flight.

VORTEX GENERATORS/

BOUNDARY LAYER ENERGIZERS

Twenty-six (26) vortex generators are installed on the top of each wing, 52 total. The vortex generators delay the drag rise at high speeds and prevent boundary layer separation to provide more effective aileron control.

NOTE

Up to 3 vortex generators may be missing for dispatch, from a total of 52 normally installed, provided the aircraft is limited to FL410.

Eleven (11) Boundary Layer Energizers (BLE) are mounted on the leading edge of each wing and all must be present for flight.

BLEs increase airflow over the wing at high angles of attack to improve stall characteristics and maintain aileron effectiveness throughout the stall regimen.

EMPENNAGE STRAKES

Strakes mounted on the empennage are designed to increase the weight and balance envelope, improve stall characteristics, and enhance lateral stability.

STATIC WICKS

Static wicks are mounted on the trailing edge of the wings and ailerons, elevators and rudder to provide proper control surface balance as well as discharge static electricity. The airplane has a total of twenty (20) static wicks installed on the wings and the empennage.

There are six static wicks located on each wing, i.e., one (1) on the tip, four (4) on the trailing edge of the wing tip, and one (1) on the trailing edge of the aileron.

Eight static wicks are located on the empennage, i.e., one (1) on the top of the rudder, two (2) on the trailing edge of the rudder, two (2) on the trailing edge of each elevator, and one (1) on the tail stinger.

Static wicks may be missing on each wing tip and/or stinger, but all others must be present for flight.

NORMAL OPERATION

PREFLIGHT

During preflight checks, the control lock should be released and all primary flight controls should be checked for proper operation and freedom of movement. All trim actuators and indicators should be checked for proper operation and takeoff settings verified. All secondary flight controls should be checked for proper operation and set properly for takeoff, i.e., flaps, speed brakes and the two-position horizontal stabilizer. The **NO TAKEOFF** annunciator should be extinguished. Nose wheel steering should be checked during taxi.

The rudder bias heater blanket is verified operational as the airplane is powered up. The **BIAS HEATER FAIL** light illuminates momentarily and extinguishes. If the light remains illuminated, dispatch is prohibited.

Before taxi, the rudder bias system is checked for proper operation by advancing each throttle separately and verify the respective rudder pedal moves forward (unequal thrust condition).

During taxi while checking the thrust reversers, verify the **RUDDER BIAS** caution light does not illuminate. Rudder bias is inoperative with the reverser(s) deployed or the EMER STOW switch(es) activated.

TAKEOFF-CLIMB

After takeoff, the gear should be allowed to complete the retraction cycle prior to retracting the flaps. If the flap handle is positioned UP while the landing gear is retracting, the gear has priority over the horizontal stabilizer (flaps will move). The stabilizer will not commence movement until the gear are up and locked. If the stabilizer does not achieve the cruise position (+1°) within 30 seconds from the time the flap handle was positioned UP , the **STAB MIS COMP** annunciator will illuminate "flashing" and trigger the MASTER CAUTION lights ON.

NOTE

Observe the position indicator to the left of the flap handle when the flap handle is moved (finger on the position indicator slot).

The yaw damper may be engaged as required after take off to enhance passenger comfort.

Engage autopilot as required.

CRUISE

The airplane may be flown manually with or without the yaw damper engaged. However, it is recommended to engage the yaw damper to provide better comfort for the passengers. Engaging the autopilot will automatically engage the yaw damper.

TURBULENT AIR PENETRATION

If severe turbulence is encountered, the following flight procedures are recommended (Refer to Section III, AFM, TURBULENT AIR PENETRATION procedures):

1. IGNITION - ON
2. Airspeed approximately 180 KIAS. Do not chase airspeed.
3. Maintain a constant attitude without chasing altitude. Avoid sudden large control movements.
4. Operation of the autopilot is recommended using basic pitch hold and lateral mode only (Disengage altitude hold).

DESCENT

Speed brakes may be used to control speed and rate of descent as required. Passenger comfort may be compromised slightly with the speed brakes extended, especially at high speeds.

APPROACH AND LANDING

When reconfiguring for approach and landing, airspeed must be 200 KIAS or below to extend the flaps to the 7 or 15° position. Speed brakes may be used to slow the aircraft if necessary.

Extend the landing gear prior to selecting flaps FULL down to prevent the gear warning horn from sounding.

The speed brakes should be retracted prior to 50 feet AGL.

If a strong crosswind landing is encountered, the rudder pedals should be neutralized to align the nose strut prior to lowering the nose gear on the runway.

Extend the speed brakes after touchdown.

AFTER LANDING

The speed brakes and flaps should be retracted after the airplane has completed its landing roll out.

SHUTDOWN

If the airplane is not expected to be towed to another location, the control lock may be engaged. The control lock should be disengaged prior to towing the airplane to protect the nose wheel steering assembly.

EMERGENCY/ ABNORMAL OPERATION

AUTOPILOT MALFUNCTION

If the autopilot malfunctions and does not respond to pilot and/or flight guidance commands, or causes abrupt flight control deviations, press and release the AP TRIM DISC switch on the control column(s) to disconnect the autopilot.

NOTE

The autopilot monitor normally detects failures and automatically disengages the autopilot.

Minimum altitude for autopilot operation:

Enroute ....1,000 feet AGL

Approaches ....180 feet AGL

Thrust Reverser Malfunctions

If thrust reverser malfunctions are experienced that necessitate placing either emergency stow switch to the EMER STOW position, rudder bias will be inoperative. However, the amber **RUDDER BIAS** annunciator will not illuminate.

Electric Elevator Runaway Trim

Attempt to deenergize the trim motor by pressing and releasing the AP TRIM DISC switch on the pilot's or copilot's control column. Reduce power as required to control airspeed. Manually trim the elevators as required. Pull the PITCH TRIM circuit breaker on the pilot's CB panel.

NOTE

Do not attempt to use the autopilot if electric trim is inoperative. The autopilot will not be able to trim out servo torque and disengaging the autopilot in this condition could result in a significant pitch upset.

Rudder Bias System Valve Not In Commanded Position (RUDDER BIAS Caution Light On)

On The Ground

Pull the Rudder Bias Circuit Breaker on the pilot's circuit breaker panel and correct the malfunction prior to flight.

In Flight

Pull the Rudder Bias Circuit Breaker on the pilot's circuit breaker panel and continue the flight in a normal manner.

NOTE

With rudder bias inoperative, rudder force and/or directional trim required for single engine operation will be significantly increased.

Rudder Bias Uncommanded Motion (LEFT or RIGHT RUDDER PEDAL Moved Forward)

UNCOMMANDED MOTION DURING GROUND OPERATION

Overpower rudder pedal deflection as required to maintain directional control. Pull the rudder bias circuit breaker on the LH circuit breaker panel and correct prior to flight.

NOTE

Uncommanded motion can only be detected with both engines at approximately the same power prior to the differential thrust condition.

Uncommanded Motion During Takeoff or In Flight

Overpower rudder pedal deflection as required to maintain directional control and climb to and/or maintain a safe altitude. Pull the rudder bias circuit breaker on the LH circuit breaker panel. The flight may be continued in a normal manner.

NOTE

Uncommanded motion can only be detected with both engines at approximately the same power prior to the unequal thrust condition. With the rudder bias inoperative, rudder force and/or directional trim required for single engine operation will be significantly increased.

Electric Trim Inoperative

If electric trim is inoperative, check the PITCH TRIM circuit breaker IN on the pilot's CB panel.

If Still Inoperative

Trim the elevators manually as required.

NOTE

Do not attempt to use the autopilot if electric trim is inoperative. The autopilot will not be able to trim out servo torque and disengaging the autopilot in this condition could result in a significant pitch upset.

Jammed Elevator Trim

Trim at Cruise Setting

If trim tab(s) jam on the elevator, maintain trim speed as long as practical until speed reduction is required for approach.

Transitioning for the approach, reduce airspeed to 200 KIAS and extend flaps to 7 degrees. After the flaps are set, extend the gear and extend flaps to 15°. Minimum airspeed for the approach is V_{APP}.

NOTE

Flaps 15°, the trim speed will be approximately 175 KIAS if the elevator trim jam occurred at V_{MO}/M_{MO}.

Prior to landing, ensure the speed brakes are retracted above 50 feet AGL and deselect the yaw damper OFF. **Multiply charted landing distance by 1.4 with flaps 15°**.

Trim at Takeoff Setting

If elevator trim tab(s) jam at takeoff setting (indicator within the takeoff range), reduce airspeed to 140 KIAS or less. Extend flaps to 15 degrees, and extend the landing gear. Maintain airspeed VAPP minimum.

Prior to landing, ensure the speed brakes are retracted above 50 feet AGL and deselect the yaw damper OFF. **Multiply charted landing distance by 1.4 with flaps 15°.**

Go-Around With Trim at Approach/Landing Setting

Jammed elevator trim at approach/landing setting during a missed approach, maintain airspeed at 140 KIAS or less and reset flaps to the previous setting when safely airborne and clear of obstacles. Land as soon as practical.

NOTE

Do not attempt to use the autopilot if the electric trim is inoperative.The autopilot will not be able to trim out servo torque and disengaging the autopilot in this condition could result in a significant pitch upset.

Use of thrust reversers in excess of idle power may increase nose up pitching force on roll out.

Multiply charted landing distance by 1.4 with flaps 15°.

Stabilizer Position Miscompare (STAB MIS COMP Caution Light On)

On Ground

If unable to extinguish the **STAB MIS COMP** annunciator prior to flight, flight is prohibited until corrected.

In Flight

If Landing

Check the flap handle in the desired detent. Use normal procedures for landing. Touch and go landings are prohibited with the stabilizer out of the landing position.

If After Takeoff

Check flap handle in the FULL UP (0°) detent. Limit airspeed to 200 KIAS and M 0.62. Do not exceed maximum altitude limit of 41,000 feet.

NOTE

Airspeed limit of 200 KIAS/M 0.62 is to prevent any possible airframe flutter from occurring with the stabilizer stuck in the landing position.

Altitude restriction of 41,000 feet maximum, applies for emergency descent observing the airspeed restriction of 200 KIAS/M 0.62.

Autopilot Out of Trim (AP ROLL or AP PITCH MISTRIM Caution Light On)

If an **AP ROLL MISTRIM** and/or **AP PITCH MISTRIM** annunciator illuminates, indicates the autopilot pitch and/or roll servos have too much torque pressure applied (airplane out of trim).

NOTE

If the **AP PITCH MISTRIM** annunciator illuminates, an amber UP or DN light on the autopilot control panel will also illuminate.

Disconnect the autopilot, if the elevator trim is not in motion.

CAUTION

BE PREPARED FOR MINOR CONTROL WHEEL FORCE REQUIRED TO MAINTAIN DESIRED FLIGHT PATH.

Adjust elevator and/or aileron trim as required. If desired, reengage the autopilot.

Autopilot Fail/Disconnect (AP OFF Caution Light On and AP FAIL Annunciation On PFD)

On Ground

Attempt reset by pulling and resetting the IC 1 circuit breaker on the copilot's CB panel. The number one Integrated Computer (IC) must be operational for the autopilot to operate.

In Flight

Press and release the AP TRIM DISC switch on the pilot's or copilot's control column. Attempt to re-engage the autopilot. If the autopilot will not reset, continue flight using manual control.

Landing With Failed Primary Flight Control Cable

Rudder

Utilize rudder trim for yaw control. After touchdown, lower the nose and deploy the speed brakes as soon as possible.

NOTE

Avoid use of thrust reversers during landing roll out. Nose wheel steering may not be available, use differential braking.

Aileron

Deselect Yaw Damper, OFF. Use rudder for directional control and limit bank angle to 15 degrees maximum. Do not use aileron trim except for gross adjustments.

If possible, choose a runway with least possible crosswind. Maximum crosswind 10 knots.

Land with flaps 15°. After touchdown, lower the nose and extend speed brakes as soon as possible. **Multiply charted landing distance by 1.4 for flaps 15°.**

Elevator

Use the manual elevator trim wheel for primary pitch control. Do not use electric trim. Make small pitch and power changes and set up landing configuration and attitude early.

After touchdown and nose wheel on the ground, extend speed brakes and apply wheel brakes as soon as possible.

WARNING

DO NOT DEPLOY THRUST REVERSERS DURING LANDING ROLL OUT.

Flaps Inoperative Approach and Landing (Not in Landing Position)

Check the FLAP CONTROL circuit breaker on the pilot's CB panel, IN. If the CB checks IN and the flaps are still inoperative, observe the following speeds during approach:

Flaps 15°, V_{APP}

Flaps 7°, V_{REF} +10 KIAS

Flaps 0° or unknown, V_{REF} +15 KIAS

NOTE

Multiply charted landing distance by 1.4

CAUTION

AVOID LANDING WITH TAILWINDS OR DOWNHILL RUNWAY GRADIENT OR AT FIELD ELEVATIONS ABOVE 6000 FEET MSL.

Angle-of-Attack System Failure (Amber AOA Annunciation On PFD)

Flaps 0° V_{REF} +15 KIAS

Flaps 7°, V_{REF} +10 KIAS

Flaps 15°, V_{APP}

CAUTION

THE STALL WARNING (STICK SHAKER) AND THE LOW AIRSPEED AWARENESS TAPE ON THE PFD WILL BE INOPERATIVE.

No Takeoff Warning (NO TAKEOFF Caution Light On and AURAL Warning)

On the ground, the elevator trim out of the takeoff range, flaps set less than 7 degrees or beyond 15°, horizontal stabilizer not at the takeoff position, and the speed brakes not fully retracted, will illuminate the **NO TAKEOFF** annunciator "steady". If power is advanced beyond approximately 85% N_2, the **NO TAKEOFF** light will commence flashing and

illuminate the MASTER CAUTION lights and an aural warning horn will sound.

If the annunciators illuminate and the aural warning sounds during the takeoff roll, abort the takeoff. Check the flaps at the proper position and elevator trim in the takeoff range. Check the speed brakes retracted (**SPD BRK EXTEND** advisory light extinguished) and horizontal stabilizer in the takeoff position (**STAB MIS COMP** annunciator extinguished).

Speed Brake(SPD BRK EXTEND Advisory Light On)

Normal indication if speed brakes are extended.

If Speed Brakes Fail to Stow

Check the SPEED BRAKE circuit breaker IN on the pilot's CB panel.

LIMITATIONS

TAKEOFF AND LANDING OPERATIONAL LIMITS

The autopilot and yaw damper must be off for takeoff and landing

Prior to takeoff, the elevator trim check in Section III, Normal Procedures, of the *AFM* must be satisfactorily completed.

Up to three vortex generators may be missing for dispatch provided the aircraft is limited to FL410 for enroute operations. There are typically a total of 52 vortex generators installed, 26 per wing.

All Boundary Layer Energizers (BLE) must be present for dispatch (11 per wing).

Rudder bias and the rudder bias heater must be operational for takeoff, and a satisfactory preflight test must be performed in accordance with Section III, Normal Procedures, in the *AFM*.

MANEUVER LIMITATIONS

No acrobatic maneuvers, including spins, are approved. No intentional stalls are permitted above 25,000 feet. Maximum maneuvering speeds are shown on Figure 2-7, in Section II, Operating Limitations, in the *AFM*.

LOAD FACTORS

In Flight

Flaps-UP, position (0°) -1.2 to +3.0g at 20,000 lbs

Flaps-T.O., TO & APPR to LAND position (7° to 35°) – 0.0 to +2.0g at 20,000 lbs

NOTE

These accelerations limit the angle-of-bank in turns and severity of pull-up maneuvers.

LANDING

Flaps Landing — 0.0 to +2.0g at 18,700 lbs

NOTE

These accelerations limit landing sink rate of 600 FPM.

Airspeed Limitations

VFE 35° (Full Flaps) ....175 KIAS

Flaps extended to 7° or 15° ....200 KIAS

Speedbrake Operating Speed VSB ....No Limit

Autopilot Operation ....305 KIAS or .75 Mach

Angle-of-Attack and Stick Shaker Systems

The angle-of-attack indicating system may be used as a reference, but does not replace the airspeed display in the PFD as a primary instrument.

The angle-of-attack system can be used as a reference for approach speed (1.3 VS1) at all airplane weights and CG locations at zero, takeoff, takeoff/approach, and landing flap positions. 1.3 V_{S1} is indicated by approximately 0.6 on the AOA gage and by the top of the white tape on the pilot's and copilot's airspeed indicators.

The angle-of-attack and stall warning system must be operable and a satisfactory preflight test must be performed in accordance with Section III of the AFM, Normal Procedures.

AVIONICS LIMITATIONS

Autopilot

1. One pilot must remain seated, with the seatbelt fastened, during all autopilot operations.

2. Autopilot operation is prohibited if any comparison monitor annunciator illuminates in flight.

3. Minimum autopilot use height for enroute is 1000 feet AGL. Minimum autopilot use height for approaches is 180 feet AGL.

TRIM

Prior to takeoff, the elevator trim check in Section III, NORMAL PROCEDURES, in the Airplane Flight Manual (AFM) must be satisfactorily completed.

QUESTIONS

1. The Ailerons are operated by:
   A. Hydraulic pressure.
   B. Mechanical inputs from the control wheels.
   C. A fly-by-wire system.
   D. An active control system that totally eliminates adverse yaw.

2. The aileron trim tab is operated by:
   A. An electrically operated trim tab motor.
   B. A hydraulically operated trim tab motor.
   C. A mechanical trim knob on the throttle center pedestal.
   D. Changing the angle of the aileron "fence."

3. Regarding the rudder:
   A. The pilot and copilot's rudder pedals are interconnected.
   B. The trim tab actuator is powered only electrically.
   C. The servo is connected to the air data computer to restrict rudder pedal deflection at high airspeeds.
   D. It is independent of the nosewheel steering on the ground.

4. Moving the flap selector lever to any position:
   A. Energizes the hydraulic system control valve closed.
   B. Energizes the flap solenoid valve to the selected position.
   C. A and B.
   D. Has no effect on the stabilizer trim.

5. The elevator:
   A. Trim tab is controlled only electrically.
   B. Runaway trim condition can be alleviated by momentarily depressing the red AP/TRIM DISC switch.
   C. Electric pitch trim has both high speed and low speed positions.
   D. Trim tab is located on the right elevator only.

6. If hydraulic power is lost:
   A. The flaps will be inoperative.
   B. The flaps will operate with the backup electrical system, but will extend and retract at a reduced rate.
   C. There is no effect on wing flap operation.
   D. A split flap condition could result if the flaps are lowered.

7. The wing flaps:
   A. If the wing flaps are positioned "UP" prior to takeoff, no visual or oral warning is present.
   B. Depend on both actuators to function to prevent a split flap condition.
   C. Can be lowered manually if electrical power is lost, but only if all hydraulic fluid has not been lost.
   D. Indirectly controls the position of the horizontal stabilizer position.

8. Regarding the gust lock:
   A. The engines may be started with it engaged.
   B. The airplane should not be towed with it engaged.
   C. It must be engaged for towing.
   D. If the airplane is towed, nosewheel steering may be damaged. It is still permissible to fly the airplane if the gear is left down.

9. If hydraulic failure occurs with the flaps extended, the flaps:
   A. May "blow upward" immediately, depending on airload if the flap handle is moved.
   B. Cannot be fully retracted.
   C. Can be retracted up electrically
   D. Flaps will remain in present position regardless if the flap handle is moved.

10. Extended speedbrakes are maintained in position by:
    A. Continuous system hydraulic pressure.
    B. Trapped fluid in the lines from the solenoid control valve.
    C. Internal locks in the actuators.
    D. External locks on the actuators.

11. The amber HYD PRESS light on the annunciator panel will illuminate during speedbrake operation:
    A. When the speedbrakes are fully extended.
    B. While the speedbrakes are extending and retracting.
    C. Both A and B.
    D. Neither A nor B.

12. A true statement concerning the speedbrakes is:
    A. The white **SPEED BRAKE EXTEND** light will illuminate whenever both sets of speedbrakes are fully extended.
    B. If DC electrical failure occurs while the speedbrakes are extended, they will remain extended since the hydraulic pressure is trapped on the extend side of the actuators.
    C. If hydraulic pressure loss should occur while the speedbrakes are extended (Hydraulic system control valve fails open), the speedbrakes will automatically blow to trail.
    D. The speedbrakes can only be retracted by placing the speedbrake switch to RETRACT.

13. If the STAB MIS COMP annunciator illuminates in flight with the flaps up:
    A. Reduce airspeed to 200 KIAS maximum and initiate a no flap landing.
    B. Reduce airspeed to 200 KIAS maximum and prepare for a normal landing.
    C. Reduce airspeed to 200 KIAS maximum and prepare for a 15° flap landing.
    D. Slow to 150 KIAS maximum and land as soon as practical.

14. Select the correct statement regarding the rudder bias system:
    A. V_{MCG} speeds are increased after rudder bias has been incorporated.
    B. V_1 speeds are decreased after rudder bias has been incorporated.
    C. Takeoff field lengths were not affected.
    D. Landing distances were decreased.

15. The rudder bias system:
    A. Will be inoperative with the thrust reversers deployed.
    B. Will be inoperative with either emergency stow switch in EMER STOW.
    C. Utilizes main system hydraulics.
    D. Both A and B above.

CHAPTER 16
AVIONICS

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ILLUSTRATIONS

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CHAPTER 16
AVIONICS

INTRODUCTION

The Citation EXCEL avionics covered in this chapter includes a Primus 1000 display and flight guidance system, emergency flight instruments, communication/navigation, pulse equipment, long-range navigation, pitot-static systems, and static discharge wicks. Avionics limitations are listed in the "Limitations" section in the back of this chapter. Many optional avionics items are available. The user should consult the applicable supplements in the AFM, Section III of the Airplane Operating Manual, and vendor handbooks for detailed information on standard and optional avionics system installed.

PRIMUS 1000 SYSTEM

GENERAL

The Primus 1000 Integrated Avionics System (IAS) is an advanced integrated system that provides display, flight director guidance, autopilot, yaw damper, and trim functions.

Standard elements consist of the following:

IC-600 or IC-615 Integrated Avionics Computers (IACs): IC-600 IACs are installed

in aircraft equipped with a standard Universal UNS-1Csp Flight Management System (FMS). IC-615 IACs are installed with an optional Honeywell FMZ FMS.

- Flight Guidance System (FGS) (Autopilot control, No. 1 IAC only)
- Electronic Flight Instrument System (EFIS)

AZ-850 air data system:

- Dual Micro Air Data Computers (MADC)

Litef LCR-93 — Attitude and Heading Reference System (AHRS)

Primus 880 weather radar

Primus II radio system

Universal Flight Management System or optional Honeywell Flight Management System.

IC-600 OR IC-615 INTEGRATED AVIONICS COMPUTERS (IAC)

At the heart of the Primus 1000 avionics system are the two IC-600 or IC-615 Integrated Avionics Computers, or IACs. Normally, Each IAC is the central processing unit for it's respective side of the avionics system, i.e. The #1 IAC processes information for the Pilot's displays and the #2 IAC handles the co-pilot's display. Except for the presence of the autopilot computer in the No. 1 IAC, the IACs are identical and interchangeable. The components common to both IACs include (Figure 16-1):

- Sensor Interface
- Flight Director Computer
- Symbol Generator

Sensor Interface — This device receives the Attitude and Heading information from the respective AHRS, as well as Altitude and Airspeed information from the respective Micro Air Data computer, and converts this data to signals usable by the IAC.

Flight Director Computer — The flight director computer is capable of taking NAV information and flight data in the IACs and converting it to a V-bar or Cross pointer guidance display on the ADI, depending on the current mode selected on the Flight Director Mode Controller. The standard configuration in the Excel is a single mode controller located on the center instrument panel utilizing only one flight director computer at any given time. Flight Director data will then be displayed as a dual synchronous display on both Primary Flight Displays (PFDs). The active Flight Director computer is selected using the FD/AP PFD1 FD/AP PFD2 Annunciator/Select switch on the center instrument panel.

Symbol Generator — This device functions as a processor, converting IAC information into a signal that can be presented on the PFDs. In the normal mode, the symbol generator in the #1 IAC is generating a display signal for both the Pilot's PFD as well as the Multifunction Display (MFD). The symbol generator in the #2 IAC is generating a display for the Co-Pilot's PFD. In the event of a malfunction or other failure, a single symbol generator can power all three display tubes. This reversionary mode is selected using the NORM – SG1 – SG2 select switch on the MFD controller.

The IAC is the focal point of information flow in the system. The two IACs are interconnected by High Level Data Link Control Lines (HDLC). This and other interconnects allow the flight guidance computers and symbol generators associated with each IAC to share, compare, and communicate large blocks of information.

The flight guidance function of the integrated avionics computer (IAC) provides digital processing of heading, navigation, and air data information to the electronic flight instrument displays (EFIS). The electronic flight instrument system displays consist of a dedicated Primary Fight Display (PFD) for each pilot and a single Multifunction Display (MFD) installed on the center instrument panel.

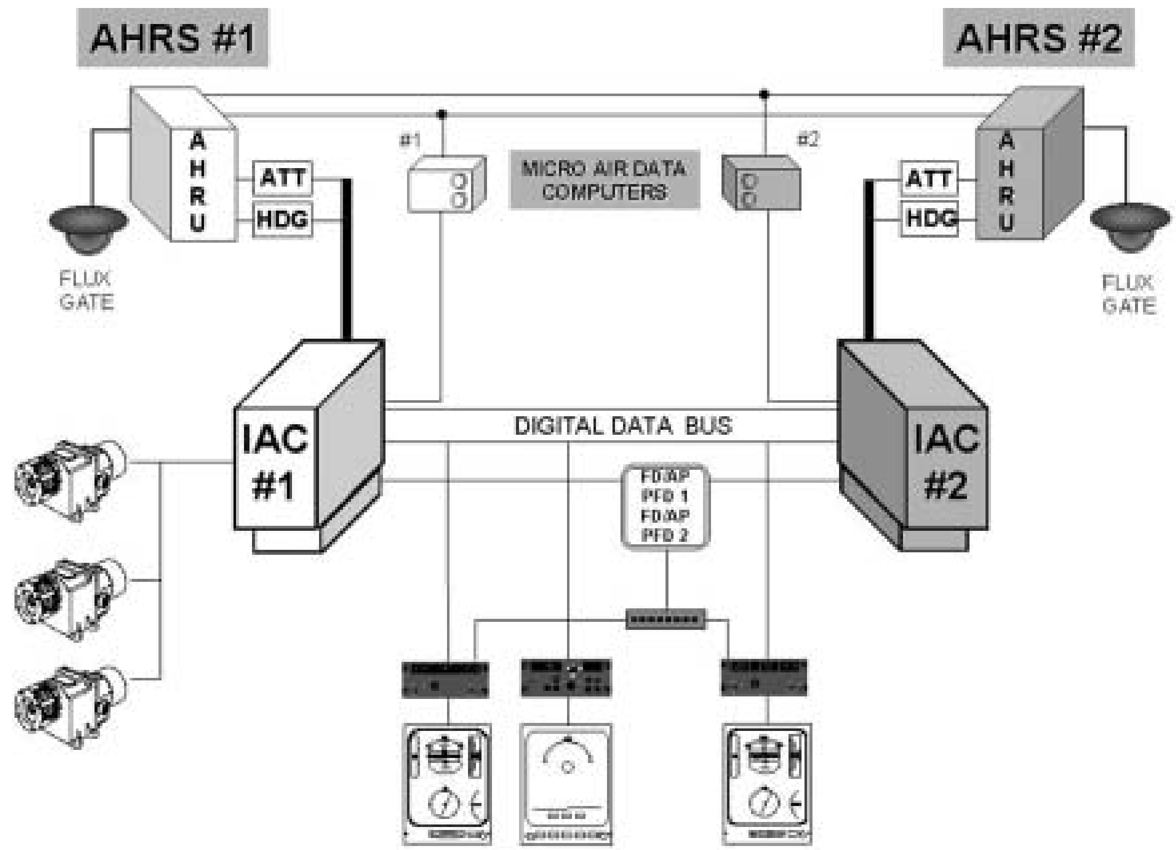


Figure 16-1 Primus 1000 System Block Diagram

The IACs convert aircraft sensor input data and information digitally to the pilot-selected formats for the Attitude Director Indicator (ADI) and Horizontal Situation Indicator (HSI) on the PFDs and data to the MFD. The IACs also process data required for the flight director command bars and steering information for the autopilot.

The IACs have a built-in multilevel test capability, which includes an automatic power-up self-test and pilot-initiated tests. It also includes on-ground maintenance testing and fault storage. The system is powered by the main DC electrical system.

SENSOR INPUT DEVICES

Attitude Heading Reference System (AHRS)

The Attitude and Heading Reference System (AHRS) is an inertial sensor installation which provides aircraft attitude, heading, and flight dynamics information to cockpit displays, flight controls, aircraft systems and instruments. The Citation EXCEL uses the Litef LRC-93 AHRS. The AHRS differs from conventional vertical and directional gyro systems in that the gyroscopic elements are Fiber Optic Gyros (no moving parts) which are

"strapped down" to the principal aircraft axes. Three micromechanical accelerometers provide rate information, and a magnetic sensor (flux gate) is used to provide long term heading references for the system.

A TAS input from the micro air data computers is used to improve the attitude reference. A digital computer mathematically integrates the rate data to obtain heading, pitch, and roll information.

The Citation EXCEL has two AHRS. The No. 1 AHRS is powered by NORMAL DC power and normally supplies data for the Pilot's side IAC and Flight Displays. The No. 2 AHRS is powered by the EMERGENCY BUS and normally provides flight information to the Co-Pilot's side IAC, Flight Displays, and the standby HSI. The AHRS system has a standby battery to provide DC power for temporary power loss, (i.e. during engine start). Should either AHRS flight data output become invalid, the other AHRS can be utilized in a reversionary mode to restore lost data.

The AHRS is made up of the following components:

- The Flux Valve — detects the relative bearing of the earth's magnetic field and is usually located in the wing or tail section away from disturbing magnetic fields.
- The Attitude Heading Reference Unit (AHRU) is the major component of the system and is composed of the following major subsystems:
- The Inertial Measurement Unit (IMU) senses the aircraft's movements, acceleration/deceleration, rotation about the aircraft axis'. It contains the fiber optic gyros, micromechanical accelerometers, and support electronics.
- The Central Processor Unit (CPU) performs the computations necessary to extract the attitude and heading information. In addition to it's computational activities, the CPU controls and monitors the operation of the entire system.
- The Input/Output Unit (I/O) supervises the handling of data between components in the system.
- The Power Supply converts aircraft power to regulated dc voltages required by the system.

The AHRS system offers advantages over conventional gimbal mounted gyros such as elimination of drift and acceleration errors. Conventional gyros are also susceptible to "gimbal lock" under certain conditions. The AHRS, operating as an inertial sensor with no moving parts, is an all attitude system and is free from such errors. The AHRS offers an improved level of system monitoring over that found in conventional systems. The central processor in the ARHU performs continuous self-checking of data and computations. A preflight test provides pilot verification of system operation through special sensor and signal path tests.

Two modes are provided for routine operation: the "normal" mode for attitude, and the "slaved" mode for heading. The "normal" mode uses true airspeed from the air data computer to compensate for acceleration induced attitude errors. The "slaved" mode uses the flux valve to align the heading outputs.

Two reversionary modes are provided to maintain performance in the event of certain types of system failures: "Basic" and "DG." The AHRS system will revert from "Normal" to "Basic" mode if the MADC TAS output becomes invalid (AHRS BASIC annunciated on the MFD). This results in an Attitude Display similar to that of a conventional vertical gyro subject to drift and acceleration errors. Occurrence of this failure is estimated to be rare, in that both MADCs are contributing TAS information to both AHRS simultaneously. Should one MADC fail, both AHRS will automatically receive TAS information from the remaining MADC.

The "DG" mode is selected by placing the "DG-SLAVE-TEST" switch from the "Slave" position to the "DG" position. This will disable the automatic slaving of the AHRS head-

ing output. Operation in this mode is similar to that of a conventional directional gyro. A two speed manual slaving input switch is provided to manually slew the heading output while operating in the "DG" mode. Although the "DG" mode may be entered at any time, the mode is usually reserved for operation in the event of a slaving failure, or for operations North of approximately 70° N Latitude where the earth's magnetic field is less reliable.

AIR DATA SYSTEM (ADS)

AZ-850 Micro Air Data Computers

The AZ-850 Micro Air Data Computer is a microprocessor based digital computer that performs digital computations, and supplies digital outputs. The MADC receives pitot-static pressures and total air temperature inputs for computing the standard air data functions. The MADC outputs data to the following components:

- Primary Flight Displays (PFDs)
- Altimeter
- Baro Set
- Mach/Airspeed Displays
- Vertical Speed Displays

The MADC also outputs data for the transponder, flight recorder, flight director, and autopilot as well as other elements in the flight control system. The MFD altitude alert knob is used to select and display the altitude reference for the altitude alerting and altitude preselect functions.

True Airspeed (TAS) Temperature Probe

A true airspeed (TAS) temperature probe (Rosemont) is located on the lower right side of the nose section. This probe is dedicated to the micro air data computers, and provides a Total Air Temperature input for the purpose of airspeed and altitude computations. The probe is anti-iced any time the master avionics switch is on and weight is off the wheels. Anti-ice electrical power is supplied by main DC power through the 15-amp TAS HTR circuit breaker located on the pilot's circuit-breaker panel.

Pitot-Static System

The Citation EXCEL is equipped with three separate and independent pitot-static systems. The two primary systems serve the pilot's and copilot's systems (Figure 16-2). The third (backup) or standby system provides pitot and static air pressure to the Standby Flight Display System (Meggitt) on the center instrument panel and to the 2-Position Horizontal Stabilizer Airspeed Switch, and it provides a source of static pressure for the cabin pressure differential pressure gage.

Pitot Tubes

Pitot pressure from the tube mounted on the left nose supplies pressure to the pilot's AZ-850 micro air data computer which, after converting the information into digital information, forwards the data to the pilot's Integrated Avionics Computer (IAC), and the Left Attitude Heading Reference System (AHRS). The pitot tube on the right nose serves the same function in the copilot's system. The standby pitot tube on the right side of the fuselage below the copilot's side window, provides pitot pressure to the Secondary Flight Display System (Meggitt), the 2-Position Horizontal Stab Airspeed Switch, and the Cabin Pressurization system.

Static Ports

Three static ports are located on each side of the airplane. The lower port on the left side and the upper port on the right side provide the static source for the pilot's system. The upper port on the left side and the lower port on the right side provide the static source for the copilot's system. The center/aft ports on each side provide static pressure for the standby pitot-static system.

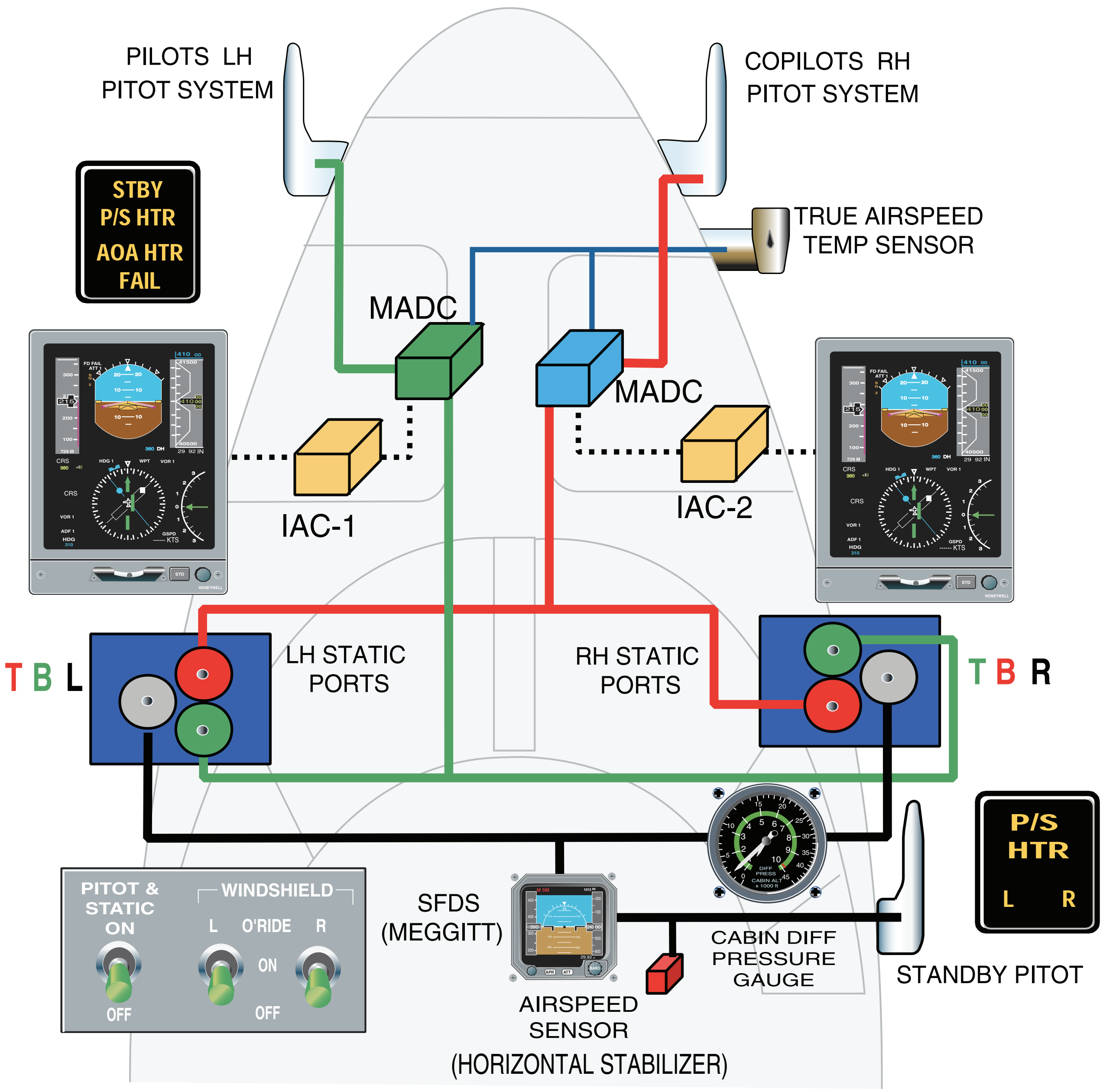


Figure 16-2 Pitot-Static System

Anti-Ice Protection

All pitot tubes and static ports are heated and controlled by the PITOT STATIC ON/OFF switch on the tilt panel. The pilot's and copilot's pitot-static anti-ice systems are powered from the main DC system through the LH PITOT STATIC and RH PITOT STATIC circuit breakers, located on the pilot's circuit-breaker panel. The backup (standby) pitot-static anti-ice system is powered from the emergency DC system through the STBY P/S HTR circuit breaker located on the pilot's circuit-breaker panel.

ELECTRONIC FLIGHT INSTRUMENT SYSTEM (EFIS)

The electronic flight instrument system (EFIS) is an integral part of the Primus 1000 integrated avionics system. The heart of each pilot's system is an IC-600/615 integrated avionics computer (IAC). In the normal configuration, the No. 1 IAC drives the pilot's PFD and MFD, the No. 2 IAC drives the copilot's PFD. The No. 2 IAC is capable of driving the MFD in a reversionary mode. Miscompare annunciations are used to indicate inaccurate information comparison between the two IACs, in critical parameters such as pitch or roll data, IAS, and BARO set.

The EFIS consists of the following elements:

- IC-600/615 Integrated Avionics Computer (IAC)
- DU-870 display units (DUs) — PFDs, and MFD
- BL-870 PFD bezel controllers (2)
- BL-871 MFD bezel controller
- DC-550 display controllers (2)
- RI-553 Remote Instrument Controller
- MC-800 MFD Controller

The EFIS displays, pitch and roll attitude, heading, course orientation, flightpath commands, weather presentations, checklists, mode and source annunciators, air data parameters, long range navigation map displays and optional TCAS and/or Enhanced GPWS or TAWS information.

EFIS brings display integration, flexibility, and redundancy to the flight control system. Essential flight information, automatic flight control, and navigation data are integrated into the pilot's prime viewing area. Selection of essential flight data, including various navigation information, aircraft performance parameters, and weather radar displays, is accomplished by using the PFD display controllers, MFD controller, weather radar controller, and the display-unit-mounted bezel controllers. Each IACs symbol generator (SG) is capable of driving all three displays. The symbol generators function as data processors for converting IAC information into a signal that is able to be displayed on the display units (PFDs and MFD).

Reversion switches allow for substituting operational sensors for failed ones, i.e., AHRS, air data computers, symbol generators, and PFD reversion to MFD.

CONTROLLERS

BL-870 PFD Bezel Controller

The PFD bezel controller is mounted on the lower front of the PFD and provides the following functions (Figure 16-3):

STD — Pushbutton returns the barometric altimeter correction to standard value (29.92 in. Hg or 1013 HPa).

BARO — Rotary set knob allows selection of reported barometric altimeter correction in either inches Hg or HPa as determined by the IN/HPA pushbutton on the Display Controller.

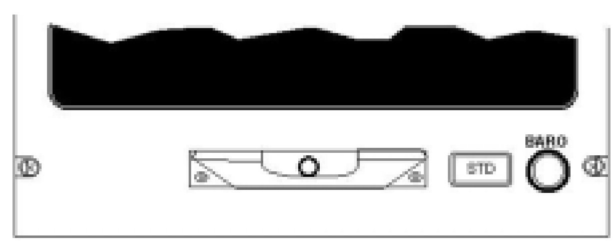


Figure 16-3 PFD Bezel Controls

NOTE

When the pilots are displaying cross-side micro air data computer (amber MADC) data on their PFDs, only the operating side MADC PFD bezel has control over both BARO settings.

The BARO set operates independently from the display controllers and does not require that the display controller be functional to set data.

BL-871 MFD Bezel Controller

The MFD bezel controller allows access for setting takeoff and/or landing V Speeds, and programming vertical navigation (VNAV) data through five menu-item pushbuttons and a rotary knob (left side) utilizing various menus (Figure 16-4).

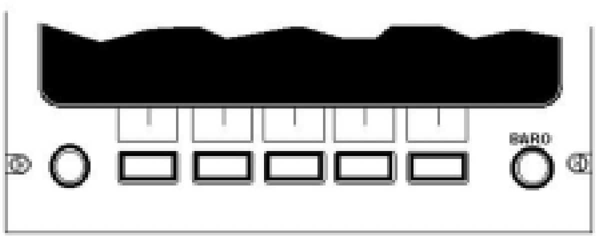


Figure 16-4 MFD Bezel Controls

MFD Menu Operation

The right rotary knob is used only for altitude preselect inputs (altitude preselect and VNAV functions). All menu pages of the MFD, display the digital readout of the selected altitude. The altitude preselect value is set in increments of 100 feet and can be changed at any time regardless of the status of any other set parameters.

DC-550 Display Controller

The display controllers, located directly to the left and right respectively of the pilot's and copilot's PFDs on the instrument panel, allow the pilots to select various formats on the PFDs. These functions are described below (Figure 16-5):

HSI Button — Controls full or WX (partial compass display). Displays 360° in FULL mode and 90° in WX (ARC) mode. Repeated activation of this switch toggles between the two displays. WX returns can be displayed on the PFD when in WX mode and WX radar is transmitting.

SC/CP Button — Selects flight director command bar display. Alternate-action toggles between single cue and cross pointer flight director display. Powerup state is single cue.

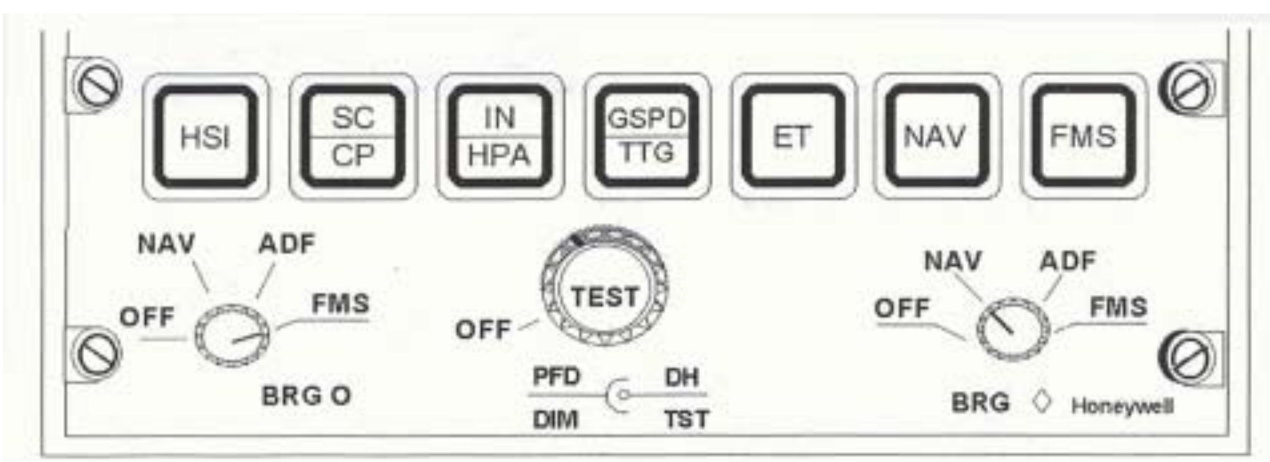


Figure 16-5 DC-550 PFD Controller

IN/HPA Button — (Inches of HG/Hecto-Pascals) Selects Barometric display mode. Pressing button toggles display between Inches of Mercury and Hecto-Pascals

GS/TTG Button — Groundspeed (GS) or time-to-go (TTG) is displayed in the lower right center of the EHSI. Pressing the GS/TTG button provides alternating selection of GS or TTG to next station or waypoint.

ET Button — Controls elapsed timer that appears in the EHSI location dedicated to GSPD/TTG. Initial actuation enters the mode at the previous position. If elapsed time is being displayed, it stops the display. Sequence of the ET button is: Reset - Elapsed Time - Stop - Repeat.

NAV Button — Pressing the NAV button selects the NAV receiver for display on the EHSI course deviation indicator (CDI). Pressing the button alternately selects NAV1 and NAV2 (annunciated VOR1 and VOR2 on the center right side of the EHSI; ILS 1 and ILS 2, if ILS frequency is tuned in NAV). The flight director interfaces with the NAV that is selected and displayed on the EHSI.

FMS Button — Selects flight management system (FMS) for display on the EHSI. The EHSI course needle represents FMS course information on the course deviation indicator. In dual FMS installations, pressing the FMS button toggles between the two FMSs for display on the PFD.

Bearing "O" Knob — This knob has four positions. The OFF position removes the No. 1 (blue) single line bearing pointer from the HSI display. In NAV position, VOR1 bearing information is displayed. In ADF position, ADF1 bearing is displayed. Selecting FMS displays bearing to the next FMS waypoint in single FMS installations, or FMS1 data in Dual FMS installations.

PFD DIM (Outer Concentric) — The DIM knob sets the brightness of the PFD. When a desired level is set, photoelectric sensors will maintain the relative brightness level in various lighting conditions. Rotating the knob to the full counterclockwise OFF position turns off the PFD, and reverts the display to the MFD as a backup, or reversionary mode in the case of a PFD failure. If both PFDs are OFF, the copilot's PFD will have priority on the MFD display. Dual reversion of both PFDs to the MFD is prohibited by FAA limitations.

Decision Height (Inner Concentric - DH) — Rotation of the DH knob adjusts the decision height display on the EADI in 5-foot increments to 200 feet and 20 foot increments above 200 feet to 990 feet. Rotating the knob fully counterclockwise removes decision height information from the display.

Test Function (TEST in Magenta) — Pressing and holding the TEST button causes the displays to enter the test mode. Flags, cautions, and all flight director and mode annunciations are tested and presented on the display. Satisfactory or unsatisfactory test results are annunciated on the display. The test also results in a self-test of the radio altimeter system; 50 feet is indicated in green in the bottom of the EADI display, and the decision height (DH) horn sounds. The TEST button is wired through a squat switch and is active only when the airplane is on the ground. The Primus 1000 test is not active in flight, but a self-test of the radio altimeter system may be made in flight if the GS capture mode is not active. The EFIS system also automatically self-tests when it is powered up, but this is normally not displayed due to the warm-up time of the EFIS tubes. If the test is not satisfactory it is so annunciated. Holding the test button for more than 5 seconds displays a ground maintenance test function on the PFD.

Bearing "diamond" Knob — This knob has three positions. The OFF position removes the No. 2 double-line bearing (white) pointer from the HSI display. In the NAV position, NAV2 bearing is displayed. In the ADF position, ADF bearing is displayed in single ADF installations, or ADF 2 bearing in Dual ADF installations. Selecting FMS displays bearing to the next FMS waypoint in single FMS

installations, or FMS 2 data in Dual FMS installations.

REVERSION SWITCHES (HDG REV/ATT REV/ADC REV)

Heading, Attitude, and Air Data Computer reversion switches are located on the pilot's and copilot's lower instrument panels (Figure 16-6).

Heading Reversion Switch (HDG REV)

The heading reversion switch is an auxiliary push-button switch which allows selection of the opposite side AHRS as an alternate (reversion) heading source for the pilot's or copilot's flight director. MAG1 or MAG2 is annunciated in amber in the center-left of both PFDs to alert the crew to the fact that both systems are utilizing the same heading source. If the AHRS/DG-SLAVE-TEST switch on the side supplying the heading information is set to DG (non-slaved mode), the MAG1(2) annunciation will be displayed as DG1(2). If there is no reversion selection and both systems are selected to their own respective sources, there is no annunciation. If there is a cross-selection on both sides, the annunciation is in amber.

Attitude Reversion Switch (ATT REV)

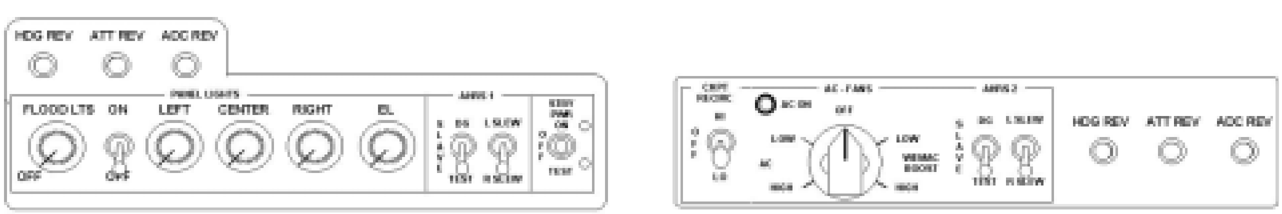
The attitude reversion switch is an auxiliary pushbutton switch which allows selection of the opposite-side AHRS as an alternate (reversion) attitude source for the pilot's or copilot's attitude indicator. ATT2 or ATT1 is annunciated in amber in the upper-left of both PFDs to alert the crew to the fact that both systems are utilizing the same attitude source. If there is no reversion selection and both systems are selected to their own respective sources, there is no annunciation. If there is a cross-selection on both sides, the annunciation is in amber.

Air Data Computer Switch (ADC REV)

The ADC reversion switch is an auxiliary pushbutton switch which allows selection of the opposite-side Micro Air Data Computer (MADC) as an alternate (reversion) Airspeed, Altitude, and Vertical Velocity Indication source for the pilot's or copilot's PFD. ADC2 or ADC1 is annunciated in amber in the upper-left of both PFDs to alert the crew to the fact that both systems are utilizing the same MADC source. If there is no reversion selection and both systems are selected to their own respective sources, there is no annunciation. If there is a cross-selection on both sides, the annunciation is in amber.

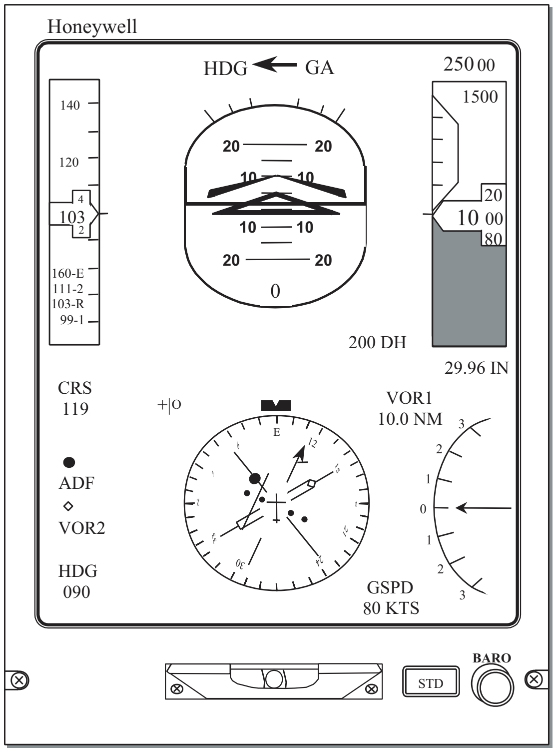
PRIMARY FLIGHT DISPLAY (PFD)

The primary flight display (PFD) on each pilot's instrument panel provides an integrated display of all pertinent flight data. The PFD is divided into the following functional sections (Figure 16-7).

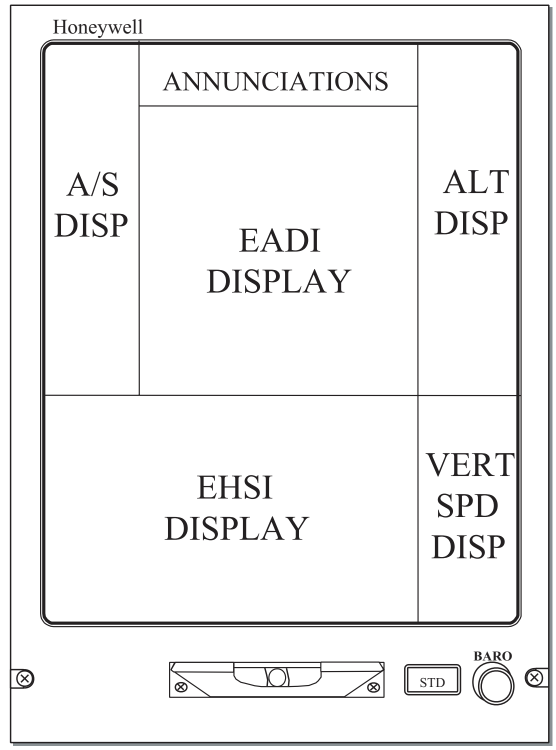


PILOT COPILOT

Figure 16-6 Reversionary Buttons



PRIMARY FLIGHT DISPLAY (PFD)



PFD FUNCTIONAL SECTIONS

Figure 16-7 Primary Flight Display (PFD)

EADI (Attitude Director Indicator)

Display — The EADI symbology uses a truncated sphere format to display standard attitude information. The attitude display receives it's input from the AHRS. When either the pitch or roll data becomes invalid, all scale markings are removed, the attitude sphere turns cyan (blue) and a red annunciation of ATT FAIL is put at the top center portion of the sphere.

Mode Annunciations — The PFD displays mode annunciations from the flight guidance processor (FPG), PFD source selection, and comparison monitor function. Lateral modes to left of center and vertical modes right of top center.

EHSI (Horizontal Situation Indicator) Display — The PFD HSI heading display receives its data from the AHRS. HSI displays include a full compass, arc, and weather modes.

Air Data Displays — The PFD airspeed, altitude, and vertical speed displays receive inputs from the MADC. Various other display data, such as radio altitude, marker beacon, and states of operation annunciations, are also displayed on the PFD.

Standard Slip-Skid Display — Standard slip-skid display is provided by the indicator on the PFD bezel controller. There is also a computed Slip/Skid indicator in the form of a Split Sky Pointer at the top of the ADI display.

Full Time Displays

Permanent displays are an integral part of the EADI portion of the PFD: the blue (sky) and brown (ground) sphere, the pitch and roll attitude reference marks, the airplane symbol, and the inclinometer which is fixed to the lower bezel. The flight director command bars are in view on power-up unless there is no lateral mode selected. The single-cue command bar is presented during initial power-up.

Various other symbols and annunciators are displayed when selected or during certain phases of flight:

Part Time Displays

Decision Height — The decision height is a three-digit display identified DH (white) in the lower-right side of the EADI presentation. The value of the decision height is identified in blue numbers. It is set by rotating the DH set knob on the display controller. Full counterclockwise rotation removes the display from view. A decision height annunciation (DH in amber inside a white box) appears in the upper left of the EADI display at radio altitudes less than or equal to the decision height setting and flashes for ten seconds. Decision height is not annunciated until it is armed. Arming occurs when the squat switch senses "in air" and a radio altitude of 100 feet greater than the selected decision height for at least five seconds.

Radio Altitude Display — The display of actual radio altitude is located in the lower part of the EADI.

Radio Altitude Rising Ground Brown Raster Display — Another indication of radio altitude is given on the barometric altitude tape. From 550 feet AGL to touchdown, a rising ground brown raster will fill the lower half of the altitude tape, displacing the normal gray raster field and altimeter scale data. The brown raster fills in the scale proportionately between 550 and 0 feet AGL.

Comparison Monitors — Amber radio altitude comparison monitor warnings (RA), attitude comparison monitor warnings (ROL, PIT, ATT), and localizer and glideslope comparison monitor warnings (LOC and GS) are located at the lower left side of the attitude display. Parameters monitored are listed as follows:

- Pitch attitude (PIT) ±5°
- Roll attitude (ROL) ± 6°
- Attitude (ATT)-Active only when both pitch and roll comparators are already tripped
- Heading (HDG) ±6°
- Altitude (ALT) ±200 feet
- Airspeed (IAS) ±5 knots
- Localizer deviation (LOC) (1/2 dot below 1,200 feet AGL)
- Glideslope deviation (GS) (1/2 dot below 1,200 feet AGL)
- ILS — Active when both LOC and GS comparators are already tripped
- Azimuth (AZ) (1/2 dot)
- Glide path (GP) (1 dot)
- MLS-Active when both AZ and GP comparators have been tripped

The comparison monitoring is active when the pilot and copilot have the same type but different sources selected for display. If, for example, the pilot and copilot both have ILS1 selected (amber annunciation of the source), no comparison monitor is active on that data (LOC, GS).

Flight Director/Autopilot Couple Arrow — The green Flight Director/Autopilot couple arrow is positioned at the top, center of the PFD. The arrow is pointing left or right to indicate which Nav display (Pilot's PFD or Co-Pilot's PFD) and Flight Director Computer, is coupled to the flight director mode controller and the autopilot. (This display is always present.)

Flight Director Mode — Annunciators Armed mode annunciations appear in white at the top left (lateral modes) and the top right (vertical modes) of the EADI presentation. Captured mode annunciations appear in green. When a mode is not selected, the annunciation is not present. As a mode transitions from armed to captured, a white box is drawn around the annunciation for five seconds.

Vertical Deviation Scale — The white vertical deviation scale appears on the right side of the attitude sphere. The driver for this scale is selected by the display controller from any one of the following sources:

- ILS glideslope
- MLS glidepath
- VNAV from the FMS or MFD bezel controller

ILS and VNAV pointers are displayed as a green rectangular box. VOR/DME VNAV deviation is displayed as a cyan pointer, and a white VNAV is displayed above the scale. The pointers are amber when both pilots select the same navigation source.

Flight Director Command Cue — The magenta flight director command cues can be selected in single-cue or cross-pointer format by pressing the SC/CP button on the display controller. In the single-cue format, if a lateral mode is not selected, the command bars remain biased out of view. Power-up default selection is single-cue.

Source Annunciations — Source annunciations, ADC1(2), ATT1(2), MAG1(2), DG1(2), SG1(2) are displayed to indicate the sources of air data, attitude, heading, and symbol generator information, respectively. If the pilot and copilot are using their normal sources, there is no source annunciated. "Cross-selections" are annunciated in amber, and when both displays are selected to the same source, the annunciation is in amber, to remind the pilots of the single source selection. Annunciation is in the upper left section of the EADI display.

Marker Beacon — Marker beacon information appears below the glide-slope indicator when ILS is tuned. A white box, in which the appropriate letter flashes when a marker beacon is passed, is located in that position when a localizer frequency is tuned on the NAV control. The outer marker is identified by a blue "O," middle marker by an amber "M," and inner marker by a white "I."

AP (Autopilot) Engage/Disengage — AP engage is annunciated by displaying AP ENG in green on the top center of the ADIs. Warning messages replace this annunciation under appropriate conditions.

TCS (Touch Control Steering) Mode Annunciator — The autopilot (AP) engage annunciator is replaced with an amber TCS annunciator when the TCS switch is pressed.

TRN KNB — Indicates the autopilot turn knob is out of the center detent (autopilot disengaged or engaged).

Category Two Approach — CAT2 (green), annunciated at the upper right of the EADI presentation, indicates that category two approach parameters have been met and the excessive deviation monitor has been enabled. A green category two approach window will be displayed around the center of the glide-slope indicator. After a CAT2 condition has been established, if any one of several conditions should go invalid (except for autopilot engaged), the green CAT2 annunciator is replaced by a flashing amber CAT2 legend which flashes for ten seconds and then goes steady. The CAT2 annunciation is removed if the autopilot is disengaged or both DHs are set above 200 feet inclusive.

MAX/MIN SPD (Maximum/Minimum Speed) Warning — When the flight director detects an overspeed condition, a MAX SPD or MIN SPD warning is displayed in amber to the left of the ADI. The warning remains annunciated as long as the overspeed or underspeed condition exists.

MAX SPD is active in FLC, VS and VNAV flight directors modes; MIN SPD is active only in VNAV mode.

EADI Caution or Failure Annunciations

Flight Director Failure — If the flight director fails, the flight director command bars disappear, and an amber FD FAIL warning appears in the top left center of the display. All FD mode annunciators will be removed.

Internal Failures — A large red X will cover the face of the primary flight display to indicate loss of signal to the display tube. A blank display tube indicates tube failure.

Radio Altimeter Failure — If the radio altimeter fails, the radio altitude readout is replaced by an amber RA. If the low altitude awareness indication is present, it will be removed.

P**ointer/Scale Failures Glide slope** (Vertical Deviation), Altitude, and Vertical Speed — Failure of pointers/scales is indicated by replacing the digital readouts with dashes, drawing a red X through the scale (IAS, ALT, GS only), and removing the pointer (GS and VS only).

Attitude Failure — Attitude failure is annunciated by appearance of ATT FAIL in red in the upper-half of the attitude sphere. The sphere will change to solid blue, and the pitch scale and roll pointer will disappear.

Excessive Attitude Declutter

The EADI display is decluttered if an unusual attitude condition is displayed. If this should occur, the following items are removed from the PFD:

- FD mode annunciations and command bars
- Marker beacons
- Vertical deviation scale, pointers, and annunciators
- ADI localizer scale
- Speed bugs and readout
- Radio altitude and DH set
- Altitude select data
- All flags and comparators except ATT and ADC (IAS/ALT)

An unusual attitude condition is defined as:

- Bank greater than 65° roll
- Pitch greater than 30° up or 20° down

PFD Electronic Horizontal Situation Indicator (EHSI)

The EHSI function of the PFD has full-time displays which are always present, part-time displays which are sometimes present, and the 90° arc compass mode.

Full Time Displays

The airplane symbol is present and provides a visual cue of airplane position relative to a selected course or heading. Other full-time presentations are similar to those on a mechanical HSI.

Heading Dial and Digital Heading Readout — Heading information is presented on standard-type compass dial format, and digital heading readout is shown above the heading dial when in the ARC mode.

Heading Select Bug and Heading Select Readout — The heading bug is positioned around a compass dial with the HEADING knob on the remote instrument controller. The bug then retains its position in relation to the dial. A digital heading select readout is provided at the lower left of the display (cyan or blue digits, white HDG label). The heading bug provides a heading error signal to the flight director.

Course Deviation Indicator — Navigation or localizer course. Course deviation and airplane position relationships are depicted as on a mechanical HSI instrument. The course deviation indicator also operates in conjunction

with the long-range NAV system. Refer to Part-Time Displays, below, for Desired Track information. The CDI is positioned by the COURSE knob on the remote instrument controller. The COURSE knob is not functional when FMS mode is selected. The CDI is magenta when FMS course information is presented, green when on-side NAV information is being presented, and yellow when off-side NAV information is being presented.

Course Pointer with Display — The course pointer rotates about the center of the arc heading display. With short-range NAV selected (VOR), the course pointer is positioned by rotating the COURSE knob located on the remote instrument controller.

TO/FROM Annunciator — Indicator points along selected course, depicting whether the course will generally take the airplane to or from the selected station or waypoint. The indicator does not appear during localizer operation.

Distance Display — Indicates nautical miles to selected station or waypoint. Distance display is in 0-399.9 NM for selection of short-range navigation equipment and 0-3999 NM format for long-range equipment. DME HOLD is indicated by an amber H next to the readout.

Navigation Source Annunciators — NAV source annunciations are displayed in the upper right corner of the EHSI presentation. Long-range sources are in magenta, and short-range sources are in green or yellow. A yellow indication means an off-side selection or that both sources are the same. The label identification will always be white. A yellow annunciation of "FMS" indicates that both pilots are selected to the FMS.

Heading Source Annunciation — Heading source is annunciated at the top left center of the EHSI presentation. A green annunciation indicates a normal selection, and amber indicates an offside selection or that both selections are the same (MAG1/MAG2 or DG1/DG2).

Heading SYNC Annunciator — The heading SYNC annunciation is located to the left of the heading source annunciation in the upper left side of the EHSI presentation. The bar in the indicator represents commands to the compass to slew in the indicated direction. Plus indicates an increase in heading, and zero indicates a reduction in heading. Slow oscillation indicates normal operation. During compass MAN (DG) modes, the annunciation is removed.

Part-Time Displays

Part-time displays are present when selected on the display controller or the flight director mode selector panel. The mode and bearing pointers available depend upon optional equipment installed and may not be, present in all installations. Some annunciations also concern other systems, which will be discussed under headings pertaining to those systems.

Bearing Pointer and Source Annunciation — The bearing pointers indicate relative bearing to the selected navaid and can be selected as desired on the display controller. Bearing pointers appear on the compass rose when they are selected by means of the knobs on the display controller, and the bearing pointer source annunciations are in the lower left of the EHSI display. If NAV source is invalid or LOC frequency is tuned, the NAV bearing pointer and the annunciation will disappear. The single-bar "O" (blue) bearing pointer is always NAV1, ADF1, or FMS1. The double-bar "diamond" (white) bearing pointer is always NAV2, ADF1 (ADF2 in dual ADF installations), FMS1 (FMS2 in dual FMS installations).

Elapsed Time Annunciation — Shows elapsed time in hours and minutes or minutes and seconds. Selection is made on the display controller.

Time-to-Go and Ground Speed — Pressing the GS/TTG button on the display controller alternates time to go (to next waypoint or navaid) and groundspeed displays.

Desired Track—When long-range navigation is selected, the course pointer becomes a desired track pointer. The long-range NAV system will position the desired track pointer. A desired track (DTRK) digital display will appear in the upper left corner of the EHSI display. When FMS is selected, the course selection knob on the remote instrument controller is inactive.

NAV Source Annunciation—Appears in the upper right side on the EHSI presentation when a NAV, ILS, or FMS source is selected as a navigation source. Distance to next waypoint or to selected VORTAC appears below the annunciation. Annunciated source will be displayed on the EHSI course deviation indicator (CDI) by changing colors.

Wind Display—The wind display (magenta direction and arrow) is located at the lower left-center of the display when FMS is selected for navigation.

Weather Radar Modes—Along the left top side of the EHSI display are the displays of the weather radar modes. These modes and displays are discussed under Weather Radar later in this chapter.

Drift Angle Bug—The drift angle bug with respect to the lubber line represents drift angle left or right of the desired track. The drift angle bug with respect to the compass card represents the aircraft's actual track. The bug is displayed as a green triangle that moves around the outside of the compass card (in either FULL or ARC mode).

FMS Waypoint (WPT) Alert—Sixty seconds prior to crossing an FMS waypoint the amber WPT annunciator is displayed to the left of the compass rose. The annunciator flashes during this time.

FMS Status Annunciation—Some critical FMS status annunciations are annunciated to the left of the compass: waypoint (WPT), offset (XTK), approach (APP), degrade (DGR), and dead reckoning (DR). XTK and APP are displayed in cyan; DR, DGR, and WPT are displayed in amber. Message (MSG) is displayed to the right of the compass in amber.

Bearing Pointer and Source Annunciations—Two bearing pointers are available: circle symbol and diamond symbol. The bearing pointers indicate bearing to the selected navaid. The pointers are selected using the display controller.

Desired Course/Track Annunciations — A desired course/track (lateral) deviation scale appears in the form of two white dots on either side of the aircraft symbol. This represents the NAV deviation from the selected source. The lateral deviation dots rotate around the center of the fixed aircraft symbol.

Wind Vector—Wind vector information is displayed in the left bottom center. The wind is shown in magenta with velocity and direction. Wind information is provided by a vector arrow showing the direction of the wind relative to the airplane symbol. The associated digital quantity indicates wind velocity.

WX/ARC Display (Partial Compass Format)

During operation in the WX/ARC mode , additional presentations are available which enhance navigation and safety of flight. Pressing the HSI button on the display controller toggles the display between the full (HSI) and partial compass (WX/ARC) display. Additional features presented in partial display are the following:

Off-Scale Arrows — In the arc mode, the heading bug and course/desired track course pointer can be rotated off the compass scale. When the HDG bug is off scale, a cyan arrow is displayed on the outer compass ring to indicate the shortest direction to its location.

Range Rings — Display of the range rings aids in the use of radar returns when WX/ARC mode is selected. The center half-range ring represents the selected radar range. The range is controlled by the weather radar controller.

Weather — Weather radar returns can be displayed on the EHSI when WX/ARC mode (HSI Button) is selected on the PFD controller. WX mode (HSI Button) forces the PFD into WX/ARC display if it was not already selected. Radar mode annunciations are presented on the upper left side of the EHSI presentation and on the lower left side of the multifunction display (MFD) as depicted in Table 16-1.

NOTE

A magenta TX is displayed in the same area when radar is transmitted but is not selected for display on the PFD.

EHSI Caution or Failure Annunciations

Amber caution annunciations appear to indicate the following situations:

DME Hold — When the DME is selected to HOLD, an amber H appears to the left of the DME readout on the EHSI.

FMS Alert Messages — Waypoint (WPT), dead reckoning (DR), or degrade (DGR) messages appear in amber at the upper center-left of the EHSI presentation to indicate, respectively, that a waypoint is being passed, the FMS is in dead reckoning, or the FMS navigation has become degraded for any of various reasons. MSG annunciated in amber at the top center-right of the EHSI display indicates that the FMS has a message on the FMS CDU.

Digital Display Cautions — When DME, groundspeed (GSPD), time-to-go (TTG), or elapsed time (ET) digital readouts fail, the digital display is replaced by dashes.

Target Alerts — An amber TGT on the left of the EHSI indicates weather radar target alert. A green TGT annunciation indicates that target mode has been selected on the weather radar.

Digital Readouts — Failure of the course or heading select signals causes these displays to be replaced by amber dashes. They are also dashed when the heading display is invalid.

Table 16-1 WEATHER RADAR ANNUNCIATIONS

OPERATING MODE	FEATURE SELECTED	DISPLAY	
		MODE ANNUN	TGT AREA
WAIT	--	WAIT (Green)-	---
STANDBY	---	STBY (Green)	---
FORCED STANDBY	---	FSBY (Green)	---
TEST	---	TEST (Green) or FAIL (Amber)	
WX	NONE VAR TGT RCT RCT/TGT	WX (Green) WX (Green WX (Green) RCT (Green) RCT (Green)	--- VAR (Amber) TGT --- TCT
FLIGHT PLAN	NONE	FPLN (Green)	---
	FPLN/TGT	FPLN (Green)	---
GMAP	NONE	GMAP (Green)	---
ANY SELECTION	VAR	GMAP (Green)	VAR (Amber)

Heading Source and Navigation Source — When both the pilot and copilot select the same heading source or NAV source, the source annunciators will be amber. If the NAV or heading sources are cross-switched, i.e., pilot to copilot and vice versa, the annunciation will also be in amber. Normal selections are not annunciated.

Heading Comparator Warning — HDG annunciated in amber at the top center left of the EHSI display indicates that the comparing system has detected an excessive difference between the two heading indicators.

Red failure annunciations appear in the following instances and locations:

Heading Failure — A heading failure results in the following indications: heading and bearing annunciations and bearing pointers disappear; HDG FAIL appears at top of heading dial; HDG, CRS SEL, and DTRK dash.

Deviation Indicator Failures — A failure in the vertical deviation or glideslope system results in removal of the applicable pointer and a red X being drawn through the scale.

Vertical Speed Display — A red X is drawn through the scale.

Air Data Displays

Air data information on the PFDs consist of airspeed, altimeter, and vertical speed displays. The micro air data computers (MADCs), fed by two independent primary pitot-static systems and a dedicated air temperature probe (Rosemont) located on the lower right side of the nose, provide data to the IACs for processing and formatting air data displays on the PFDs.

Airspeed Displays — The airspeed section of the PFD display is to the left of the ADI display. The display consists of a "rolling digit" window in the center of an airspeed vertical tape. The resolution in the window is in 1-knot intervals. The moving vertical tape moves behind the window and displays airspeed at 20-knot intervals. The tape rolls downward; larger numbers roll down from the top of the scale. The range of speed is 40 to 400 knots with tick marks at 10-knot intervals.

Trend Vectors — An airspeed trend vector (magenta), which displays an indication of the direction and rate of airspeed change, extends vertically from the apex of the current airspeed value display window. It extends upward for acceleration and downward for deceleration. The trend vector represents a prediction of what the airspeed will be in ten seconds if the current change in airspeed is maintained.

V-Speed Indications — Bugs for six V speeds are provided to allow pilot selection of key airspeeds by means of the multifunction display (MFD) bezel buttons. They are labeled "1" (V_1), "R" (V_R), "2" (V_2), and "E" (V_{ENR}) (this airspeed is automatically displayed whenever V_1, V_R, or V_2 is selected for display) and "RF" (V_{REF}) and "AP" (V_{APP}). When the takeoff speeds are selected, digital indications appear at the bottom of the airspeed display, as well as the bugs being placed into position. The bugs are positioned on the right outside edge of the airspeed tape. They consist of a horizontal T-shaped symbol with its respective label positioned to the right of the symbol. All the takeoff set bugs are removed from the display when the airplane airspeed exceeds 230 knots, and the landing speed bugs are removed when power is turned off.

When the airspeed is below 40 knots, V_1, V_R, V_2, and V_{ENR} are displayed in the bottom portion of the airspeed tape in the form of a digital readout. The digital readout of the set value is displayed along with the bug symbol and are labeled in ascending order, starting with V1. Upon power-up, the digital readouts for the set bugs are amber dashes. As the V-speeds are set on the MFD menu, the digital readouts follow the readout on the MFD and set accordingly. The digital readouts are removed from the display at weight-off-wheels.

Standby Airspeed — Standby airspeed indications are always available from the

Secondary Flight Display System (Meggitt Tube), which is discussed later in this chapter under Emergency Flight Instruments.

Overspeed Indications — Below 8,000 feet altitude the limiting airspeed (V_{MO}) is 260 KIAS; between 8,000 and 26,515 feet the limiting airspeed is 305 KIAS. When one of these limits is exceeded, the airspeed indication in the window is changed to red and an amber annunciation. Also, to the left of the attitude sphere MAX AIRSPEED is illuminated . A red thermometer-type tape is also presented on the inside of the airspeed scale. The thermometer extends from V_{MO}/M_{MO} to larger airspeeds on the tape and appears in the indication as the airspeed reaches into the range near V_{MO}/M_{MO}. When the limiting airspeed is exceeded, the overspeed warning horn sounds and continues to sound until the airspeed is reduced below the limit speed.

Low Airspeed Awareness — A red, amber, and white thermometer-type display located on the inside of the airspeed scale gives indication of low airspeed as calculated by the AOA input. The white extends from 1.3 to 1.2 V_{S1}, the amber band extends from 1.2 to 1.1 V_{S1} (approximately stick shaker speed), and the red extends from stick shaker speed to the smaller airspeeds on the tape.

Mach Number Display — A digital readout of indicated Mach number is displayed below the airspeed scale. The Mach number comes up on the display when Mach exceeds 0.390 and is removed when it falls below 0.380 Mach. Resolution of the Mach display is 0.01 Mach.

Altitude Indications

The altitude display is located to the right of the EADI. The altitude tape is a moving scale display with a fixed pointer (center of window). The scale markings on the tape are labeled in 100-foot increments. The scale tape displays larger numbers at the top.

The range of altitude window is from -1,000 to 60,000 feet with tick marks located at 500-foot increments. The scale is labeled in 500-foot intervals, and single-line chevrons are located at each 500-foot increment. Double-line chevrons are located at each 1,000-foot increment. The chevrons extend back to the approximate midpoint of the altitude tape and are connected with each other by a vertical line. The left side of the “rolling digit” window has the same angle as the chevrons.

Altitude Digital Display — A digital display (green) of the actual altitude value is contained in the display window. This data is a magnification of the digits on the scale and is readable to within a 20 foot resolution. The digits within the pointer scale are white. For climb/descent rates greater than 1,000 feet per minute, the rolling drum digits are replaced by two dashes to enhance altitude scale readability. Below 10,000 feet, boxed hash marks are used to show that the ten-thousand-foot digit is missing.

Altitude Alert Select Display — Altitude alert select data is displayed at the top of the altitude scale. This data is set by using the right-side MFD bezel set knob.

The altitude preselect data is cyan (blue). When the aircraft is within the altitude alert region (±1,000 feet), the box and the set data turn amber. When a departure from the selected altitude capture occurs, the select data also changes back to amber. When the aircraft approaches the set altitude, within 250 feet, the box and the altitude data turns back to cyan (blue). A momentary audio alert sounds when the aircraft is 1,000 feet from the preselected altitude or has departed 250 feet from the select altitude after capture.

Altitude Select Bug — The cyan (blue) altitude select bug travels along the left side of the altitude tape. The altitude select bug is notched to fit the 1,000- or 500-foot altitude tape chevron format. The bug appears on the scale across from the altitude value set in the altitude alert select display. If the bug is moved off the current scale range, half of the bug remains on the scale to indicate the direction to the set bug.

Low Altitude Awareness — At radio altitudes of 550 feet or less, the lower part of the altitude tape linearly changes from a gray raster to brown and the altimeter scale markings are removed. At zero radio altitude, the brown raster touches the altimeter reference line.

Barometric Altimeter Setting — The baro set window is located directly below the altitude tape. The pilot has the ability to set the altimeter in either inches of mercury (in. HG) or hectopascals (HPa) as selected with the PFD display controller. If the on-side display controller is invalid, the SG defaults to the last selection (IN or HPa). The baro set data is always cyan (blue).

Altitude Trend Vector — The magenta altitude trend vector is displayed on the left edge of the altitude tape and provides an indication of the rate of altitude change. The trend vector extends vertically from the apex of the current altitude display window. The vector extends up for positive vertical trends and down for negative values. The vector represents a prediction of what the altitude will be in ten seconds if the current vertical speed is maintained. The MADC outputs altitude rate of change.

Standby Altitude — Standby altitude indications are always available from the Secondary Flight Display System (Meggitt Tube), which is discussed later in this chapter under Emergency Flight Instruments.

Vertical Speed Display

The vertical speed display is located to the right of the EHSI and directly below the altitude display. Vertical speed data is developed in the micro air data computers, which sense the rate of change of altitude from inputs of the static system. The computers convert the data into digital form and transmit it through the digital data bus system to the IC-600/615 display guidance computers, which forward it to the DU-870 primary flight displays (PFDs), where it is generated into a visual display.

VS (Vertical Speed) Analog Scale — The VS scale is a fixed arc scale with moving pointer. The scale on the display ranges from +3,500 to -3,500 feet per minute. Display scale markings are 0, 1, 2, and 3. The scale and its marking are white.

VS Digital Display — A digital display of the actual VS value is located in a box, on the zero reference line. This data is a magnification of the digits on the scale and readable to a 50-foot-per-minute resolution. The digits within the box are green. Maximum value is 9,900 feet per minute. For values between ±500 feet per minute, the digital display is removed. At values beyond +/-500 feet per minute the digital value of vertical speed is displayed.

For vertical speeds greater than ± 3,500 feet per minute, the pointer is positioned in the appropriate direction at the end of the scale. The digital display shows the actual vertical speed value.

Flight Director VS Target Display and Bug — Engaging the vertical speed mode brings the VS target bug into view. The VS target bug moves along the right side of the VS scale. The bug lines up with the value on the VS scale that is set with the autopilot controller pitch wheel or TCS button. The bug is always cyan (blue). The digital readout of the target is displayed on top of this vertical speed scale. The target speed comes from the flight guidance system.

TCAS II Resolution Advisory Display (Option) — The TCASII system displays a green “fly to” target and a red “do-not-fly” band on the vertical speed display that commands the pilot to comply with a resolution advisory (RA) to avoid a potential aircraft conflict.

TCAS Status Message — The TCAS status messages are presented to the top left of the vertical speed display. When a TCAS II RA is displayed, the vertical speed digital display notches the color of the red or green band where the pointer is located.

MULTIFUNCTION DISPLAY SYSTEM (MFD)

The multifunction display (MFD), the center cathode ray tube (Figure 16-8), serves as the weather radar indicator. It can be used to display the horizontal navigation situation, either short range (VORTAC) or long range (FMS), and to display electronic checklists. It also provides backup capability to the EFIS systems.

If a symbol generator on one side fails, the pilot can, through the MFD controller, select the opposite-side symbol generator to take over the failed side's display, and operation of the EFIS in that position will continue as before, with the selected symbol generator powering all three displays.

The multifunction display system expands on the navigation mapping capability of the EFIS, especially in conjunction with the flight management system (FMS). The MFD display may be used independently for navigation and mapping information without disturbing the EHSIs, which then may be used without additional displays which would result in more "clutter" on the EHSI. The weather radar display may be selected independently or overlaid on the navigation display provided by the flight management system, in order to show the airplane route with respect to the displayed weather returns.

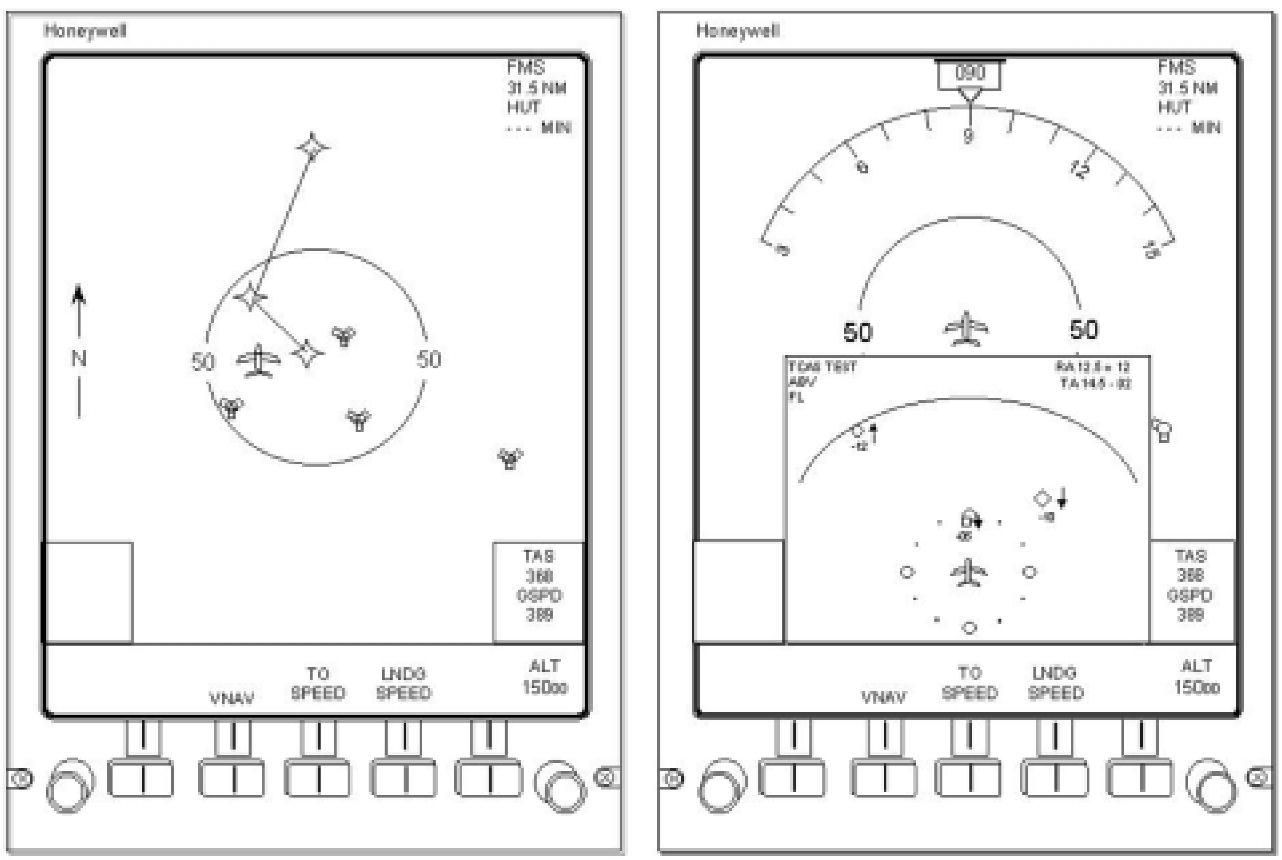


Figure 16-8 MFD

Multifunction Display Controller

The MFD controller, located on the center instrument panel to the right of the MFD (Figure 16-9), allows mode selections, display control, and symbol generator reversion control of the pilot's and copilot's systems. In addition to its navigation, reversion, and checklist functions, the MFD control also provides for control of the display of the optional traffic alert and collision avoidance system (TCAS) (Figure 16-8).

MFD Modes of Operation

The modes of operation available to the MFD system are listed as follows:

MAP Mode

The MAP function is a partial-arc, heading-up display which is selected by the alternate-action MAP/PLAN pushbutton. The MFD display cycles from MAP to PLAN as the MAP/PLAN button is pressed. The MAP format allows totally independent use of the MFD display for navigation mapping and allows increasing the maximum range, beyond normal radar range, on the display which normally serves as the radar indicator. Power-up mode is the MAP mode. To add weather to the display, press the WX button on the MFD controller.

The MAP format is always oriented to the airplane heading, and the airplane symbol is located at the center of the display. When coupled to the FMS, the NAV route, with up to ten waypoints, can be displayed to the range limit. When weather returns are selected, range control defaults to the weather radar controller.

PLAN Mode

In PLAN mode, the top of the display is oriented to True North; a three-inch range is displayed and centered horizontally on the displayed area. An aircraft symbol is plotted at present position (if present position is on the display) and is oriented with respect to heading. The PLAN mode display encompasses 360°. Weather radar returns cannot be presented in the PLAN mode.

TCAS (Optional)

The TCAS button is optional and is used to manually select TCAS traffic display on the MFD.

Weather (WX) Mode

The WX mode allows the MFD display to be used as a weather radar indicator. In WX mode, weather data is presented on the MFD and is superimposed upon the normal navigation display. Weather radar can be selected for display on the MFD only if MAP mode is selected. If the MFD is in PLAN mode, selection of WX mode forces the display into MAP mode. Range selection is controlled by the weather radar controller. When the WX button is tog-

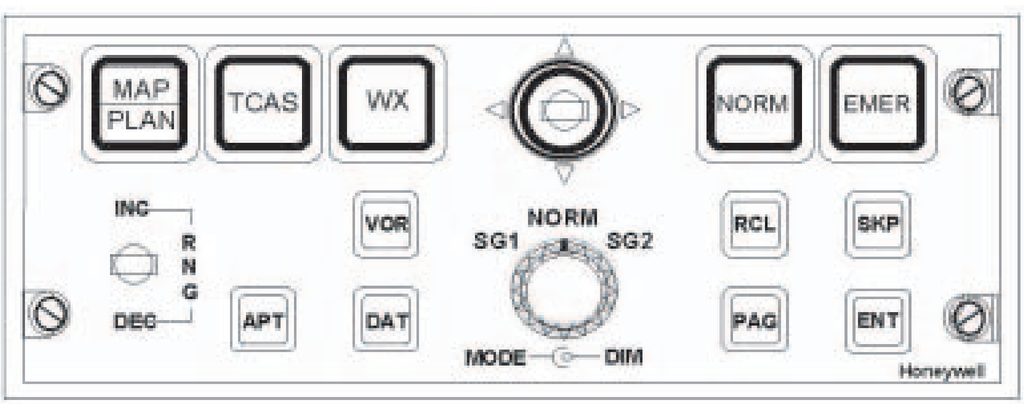


Figure 16-9 MFD Controller

gled, the progression of selection is: WX on, WX off. Annunciation of weather modes, warnings, and antenna angle are provided at the lower middle left of the MFD display. Annunciations are color-coded in magenta, green, and amber according to the importance of the display. Operation of the weather radar is discussed later in this chapter.

Checklist Modes

The NORM button on the controller provides a display of flight plan waypoints or entry into the normal checklist display function. The normal checklists are arranged in the order of standard flight operations. Button actuations cause presentation of the normal checklist index page that contains the lowest order incomplete and unskipped checklist with the active selection at that checklist.

The RCL, SKP, PAG, and ENT buttons and the joystick provide control of this function and are discussed under MFD Controls below.

The EMER button on the controller provides entry into the emergency checklist display function. Actuation of EMER results in the presentation of the first page of the emergency checklist index with the active selection at the first checklist. The RCL, SKP, PAG, and ENT buttons and the joystick provide control of this function and are described in MFD Controls below. These controls perform as described for NORM with the exception of the action taken upon completion of the checklist. All checklist items are removed from the page, and "EMERGENCY PROCEDURE COMPLETE" is written below the amber checklist title. This will be cleared when the index is selected. The SKP, PAG, and ENT buttons will be inoperative.

EFIS Backup Modes

In case of a symbol generator failure, the select knob may be selected to the opposite side SG. If SG1 is selected, the pilot's symbol generator is driving all three PFD displays. SG2 means the copilot's symbol generator is driving all three PFD displays. In these cases the MFD is normal, and both PFD displays have the same format. The multifunction display has no complete symbol generator function of its own.

Traffic Collision Avoidance System II (TCAS II) (Optional)

The TCAS mode allows the TCAS window to be displayed when TCAS is installed in the airplane. The TCAS resolution advisory is displayed on the PFD, and traffic advisories are displayed on the MFD.

MFD Controls

Dim — This knob controls overall MFD CRT dimming in addition to the automatic dimming feature accomplished by CRT-mounted photodiodes. Turning the knob counterclockwise dims the display. The WX display is dimmed at the same time.

Joystick — The function of the joystick depends upon the type of MFD display:

- MAP or PLAN — Moves the designator in directions shown.
- TEXT — Vertical actuations — Acts as a cursor control by changing the active line. This provides an additional means of skipping lines or returning to a previously skipped line.
- Horizontal Actuations — Controls paging. Actuation to the right increases the page number, and actuation to the left decreases the page number.

MAP/PLAN — Pressing the MAP/PLAN button selects the MAP MFD display mode. Pressing it again selects north-up PLAN mode.

WX — Weather radar data may be displayed with the MAP mode. The toggling sequence of this button is: WX on, WX off. If PLAN mode is selected, selection of MAP mode will be forced when WX mode is selected.

VOR — The VOR button is used to display up to four of the closest VORs, with identifiers, that are not on the active flight plan list, on the MFD MAP and PLAN displays. Pressing the

button a second time will remove the identifiers. A third time will remove the VORs.

APT — The APT button is used to display up to four of the closest airports, with identifiers, that are not on the active flight plan list, on the MFD MAP and PLAN displays. Pressing the button a second time will remove the identifiers. A third time will remove the Airports.

DAT — This button is used to add identifiers to the long-range NAV displays on the MFD MAP and PLAN displays.

Range controls (INC and DEC)—The MFD range controls are active only when WX is not selected display. Selectable ranges are 5, 10, 25, 50, 100, 200, 300, 600, and 1200 NM. The INC switch position increases the selected range, and the DEC position decreases the selected range.

NORM — When this button is pressed, the MFD displays the index page containing the lowest numbered uncompleted or unskipped checklist with the active line at that checklist. All waypoints of the current flight plan may be displayed.

While operating in this mode, as a checklist is completed, the system automatically steps to the next uncompleted procedure of the index.

EMER — Actuation results in the display of the first page of the emergency checklist index.

RCL — The function of this button depends upon the type of MFD display:

- MAP or PLAN — Recalls the designator to its home position.
- TEXT — Recalls the lowest numbered skipped line in a checklist by changing the active page and/or line.

SKP — The function of this button depends upon the type of MFD display:

- MAP or PLAN — Skips the designator to the next waypoint. If the designator is not at the home position, the displacement line is moved to the next waypoint.
- TEXT — Actuation skips the active line in a checklist or index and advances the active selection to the subsequent line. If the line skipped is the last line, the active selection reverts to the lowest numbered skipped line.

PAG —Actuation advances the page count and places the active line selection at the first line of the page. Actuation with the last page displayed results in display of the lowest numbered page containing a skipped line with the active line selection at the lowest numbered skipped line.

ENT — The function of this button depends upon the type of MFD display:

- MAP or PLAN — With the designator moved from its home position, actuation of these buttons enters the designator LAT/LOG as a waypoint in place of the TO waypoint.
- TEXT — Actuation checks off a line in a checklist or selects an index line item for display.

Auxiliary EFIS Annunciators

Indications are located in the upper left of the multifunction display.

- IC-1 HOT — Indicates overtemperature condition of pilot's IC-600/615 display guidance computer.
- IC-2 HOT — Indicates overtemperature condition of copilot's IC-600/615 display guidance computer.
- IC-1-2 HOT — Indicates overtemperature condition of both IC-600/615 display guidance computers.
- IC-1 FAN — Indicates failure of pilot's IC-600/615 cooling fan.
- IC-2 FAN — Indicates failure of copilot's IC-600/615 cooling fan.

- IC-1-2 FAN — Indicates failure of both IC-600/615 cooling fans.
- CHK PFD1 — IC-600/615 display guidance computer detects a wraparound failure in PFD1. Data displayed is not being updated. Verify critical data with other flight instruments. Comparator warnings may not be active.
- CHK PFD2 — IC-600/615 display guidance computer detects a wraparound failure in PFD2. Data displayed is not being updated. Verify critical data with other flight instruments. Comparator warnings may not be active.
- CHK PFD1-2 — IC-600/615 display guidance computers detect a wraparound failure in both PFDs. Data displayed is not being updated. Verify critical data with other flight instruments. Comparator warnings may not be active.

Flight Director Mode Selector

The Flight Director (F/D) mode selector consists of seven push-on, push-off switches that select various flight director/autopilot modes of operation (Figure 16-10). The green mode activation light in the switch (button) is illuminated if the corresponding mode is in the arm or capture state. The status of the selected mode is displayed in white letters (annunciations) in the primary flight display (PFD) when armed, and in green when capture has occurred.

The flight director can be selected off by deselecting all of the modes on the flight director mode selector. The command bars will bias out of view. If single-cue flight director operations is selected on the DC-550 display controller, the flight director/autopilot will not engage if only a vertical mode is selected. If no modes are selected on the flight director mode selector, the autopilot will engage in a basic heading-hold/pitch-hold mode.

Operation of the various modes is explained later under PRIMUS 1000, Flight Director Modes. The pilot and copilot may select either NAV1 or NAV2 for display on their respective primary flight display (PFD) by means of the NAV button on the display controller. The respective on-side NAV is automatically selected upon power-up. The selection of NAV1, NAV2, or FMS is annunciated in the upper right corner of the PFD as VOR1, VOR2, and FMS respectively.

The selection of NAV1, NAV2, or FMS on the display controller pushbuttons controls the source of navigation information to the flight director, as well as selects the source of navigation information displayed on the EHSI course deviation indicator (CDI) of the PFD. A switch (AP/FD PFD1 - AP/FD PFD2), located to the right side of the Mode Controller, is installed to determine which flight director computer/PFD controls the autopilot. The position of this switch can be changed with the autopilot engaged or disengaged, however, the Flight director modes will drop out and the autopilot will revert to basic modes if engaged.

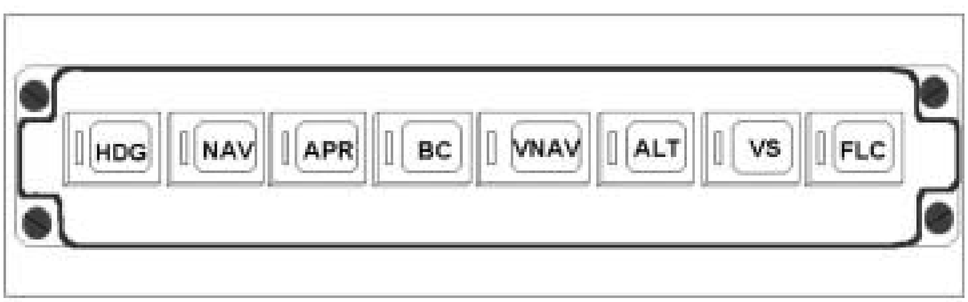


Figure 16-10 Flight Director Mode Controller

AUTOPILOT CONTROL PANEL

The autopilot control panel, mounted on the pedestal, provides the means of engaging the autopilot and yaw damper, as well as manually controlling the autopilot through the turn knob and pitch wheel (Figure 16-11).

The autopilot (AP) engage switch is used to engage the autopilot and yaw damper. The yaw damper (YD) switch is used to engage and disengage the yaw damper without the autopilot. Use of the yaw damper while manually controlling the airplane aids in airplane stability and passenger comfort. The push-on/push-off AP and YD switches are illuminated green when engaged. Pressing the AP switch when the autopilot is engaged disengages the autopilot but leaves the yaw damper engaged. Pressing the YD switch when both yaw damper and autopilot are engaged turns off both the yaw damper and the autopilot. The yaw damper and autopilot may also be disengaged with the red AP TRIM DISC button on the pilot's and copilot's control wheels. Pressing the go-around (GA) button on either throttle, disconnects the autopilot and forces the flight director into the go-around mode; the yaw damper remains engaged.

The pitch wheel allows manual pitch control of the airplane proportional to the rotation of the wheel and in the direction of wheel movement. Movement of the wheel also cancels any other previously selected vertical mode.

The turn knob allows manual bank control of the airplane proportional to and in the direction of knob movement. Turns with a maximum bank angle of 30° can be performed with the turn knob. The turn knob must be in the center detent position before the autopilot can be engaged. Rotation of the turn knob out of detent cancels any other previously selected lateral mode.

The elevator trim indicator shows an out-of-trim condition, in the direction indicated by illumination of UP or DN in the TRIM annunciator, when a sustained trim input is being applied to the elevator servo. The indicator should be OFF before engaging the autopilot. If the TRIM annunciator is illuminated the autopilot should be disengaged, the pilot should be prepared for an out-of-trim condition in the annunciated direction. A separate additional **AP PITCH MISTRIM/AP ROLL MISTRIM** annunciator is located on the annunciator panel, where it is more readily visible to the pilots. The **AP PITCH MIS-TRIM** annunciator is a repeat of the TRIM annunciator on the autopilot control panel.

The AP ROLL MIS-TRIM annunciator indicates to the pilot that a sufficient level of roll mis-trim is present and the pilot must be prepared for an out-of-trim roll condition if the autopilot is disconnected.

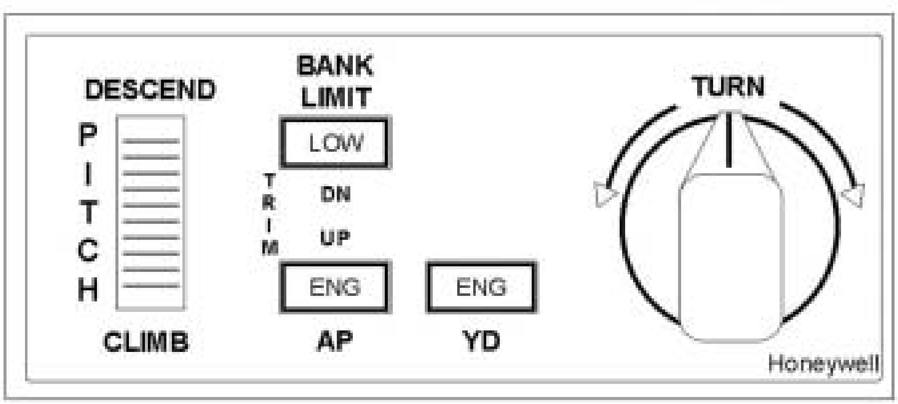


Figure 16-11 Autopilot Control Panel

The bank limit (LOW) mode may be selected if it is desired to limit the maximum bank angle during autopilot operation. The mode is limited to use in conjunction with heading (HDG) mode only. When the bank limit mode is engaged, the autopilot maximum bank angle is limited to 14°. When the mode is engaged, LOW annunciates in the pushbutton. Low bank mode is automatically selected when climbing through 34,000 feet altitude, and automatically canceled when descending through 33,750 feet. If heading mode is selected and then deselected while low bank is engaged, low bank mode is disengaged and the engage light extinguishes during the time heading mode is disengaged, but low bank mode re-engages and the LOW annunciator re-illuminates when heading mode is re-engaged.

The autopilot is normally disengaged in one of three ways: (1) depressing the AP TRIM DISC switch on either yoke, (2) electrically trimming the elevator trim system, or (3) depressing the go-around button on either throttle (if the F/D is engaged). Actuation of the touch control steering button interrupts the pitch and roll servos until the switch is released; the yaw damper remains engaged. If the autopilot is disengaged by any of the above three ways, a warning tone sounds for one second, and the amber AUTOPILOT OFF light illuminates for one second. Any other disconnect causes the warning horn to sound for one second and the AUTOPILOT OFF light to stay illuminated. The amber light can be extinguished by holding the AP TRIM DISC switch for two seconds, or by pressing the electric trim switch or the go-around (GA) button on either throttle. The autopilot also disengages if an overriding force (sustained torque) is applied to the vertical or horizontal axis for a minimum preset time. Disconnect is annunciated by the one-second disconnect tone and illumination of the autopilot disconnect light until the light is extinguished by one of the above methods.

PRIMUS 1000 INTEGRATED OPERATION (EFIS/FLIGHT DIRECTOR/AUTOPILOT)

The Primus 1000 system in the Citation EXCEL operates through displays of the pilot's (or copilot's) electronic flight instrument system (EFIS). The autopilot and EFIS systems are integrated, and unnecessary system redundancy has been eliminated. The result is an overall simplification over previous systems and greatly simplified interface requirements for the flight director function. If a particular EFIS unit is operational, the flight director will also be operational, and conversely if the EFIS has failed, the flight director will also fail. The display is available as a single-cue or a double-cue (cross pointer) presentation, the selection is made by using the SC/CP button on the display controller. The presentation on initial power up is single-cue. Glide-slope and VNAV vertical path information are presented on the right side of the electronic attitude director indicator (EADI) section of the primary flight display (PFD). The pertinent command bar(s) of the flight director can be brought into view, when double-cue display is selected, by selecting any mode. If single-cue mode is selected, selection of a vertical mode only will not bring the command bars into view.

The autopilot may be switched to the pilot's PFD 1 or the copilot's PFD 2 by means of an illuminated selector switch (AP/FD PFDI – AP/FD PFD2) located on the center instrument panel. This switch determines which flight director / PFD NAV display provides guidance to the autopilot.

The Primus 1000 system incorporates a wide variety of capabilities that produces one of the most precise, flexible, and easy-to-use systems in airplanes today. The flight director and autopilot can be used independently or together. The airplane may be flown manually, using the guidance provided by the modes selected on the flight director, or when the autopilot is engaged and coupled to the flight director, it controls the airplane using com-

mands generated by the flight director computer. Disengagement of the autopilot has no effect on the FD modes in operation at the moment of disengagement, except when using the go-around button, in which case a wings-level 10° nose-up attitude is commanded and all other FD modes are reset.

When the autopilot is engaged without a mode selected, manual pitch and roll commands may be made by use of the turn knob and pitch wheel on the autopilot controller. Touch control steering (TCS) can be used to maneuver the airplane or to modify the commands to the FD and AP. If the autopilot is not engaged, the TCS button can be used to synchronize the command bars to the airplane attitude. If HDG mode has been selected, BANK LIMIT mode may be engaged, and the maximum bank angle is limited to approximately 14°.

Basic Autopilot

The basic autopilot, without any inputs from the flight director system, can be used for pitch, roll, and heading hold. The autopilot holds the pitch attitude existing at the moment of AP engagement and the pitch attitude existing at the moment of disengagement of a vertical mode.

The autopilot can be engaged in any reasonable attitude; however, unless touch control steering (TCS) is used in conjunction with autopilot engagement, the autopilot rolls wings level if engaged while in a bank. If the bank is less than 6° at engagement, the autopilot holds the heading indicated when the autopilot is engaged. If the bank is over 6° at engagement, it holds the heading indicated when the airplane rolls through 6° of bank on the way to wings level. If a lateral mode is disengaged, the autopilot holds the heading existing at the moment of disengagement. If the turn controller is out of the center detent position, the autopilot will not engage (annunciated in amber on the PFDs).

Touch Control Steering (TCS)

Touch control steering (TCS) enables the airplane to be maneuvered manually during autopilot operation without cancellation of any selected flight director modes. To use touch control steering, press the TCS button, maneuver the airplane, and release the TCS button and the A/P reengages. TCS is operable with all autopilot modes. During TCS operation the yaw damper remains engaged.

If the autopilot is engaged and it is desired to hold a bank angle, roll into the bank desired, press the TCS button and release, the A/P holds the bank angle upon release. The bank is maintained if it is in excess of 6°. The airplane may be rolled level with the turn knob. The memory function holding the autopilot in a bank is canceled when the knob is moved out of detent.

Operating in speed (SPD) (IAS or MACH annunciated) mode, vertical speed (VS) mode, or altitude hold (ALT) mode, the TCS button may be depressed and the airplane maneuvered to a new reference. When the TCS button is released, the flight director/autopilot maintains the new reference.

Pitch Synchronization

Flying the airplane manually and using the flight director, the command bar may be matched to the existing pitch attitude by pressing the TCS button (command bar assumes a neutral position) and releasing it; the command bar synchronizes to the airplane attitude at the moment of release.

Flight Director Modes

Heading

The heading mode (HDG-annunciated in green letters in the top right of the EADI) can be used with the flight director (FD) only, or in conjunction with the autopilot. When the heading (HDG) mode is selected on the FD mode selector, the command bars come into view and display a steering command (HDG

cursor bug) on the PFD, and controlled by the remote instrument controller on the center pedestal. The command bars synchronize vertically to the pitch attitude at the time of HDG selection. Heading mode is engaged automatically if another lateral mode is selected and the airplane is outside the capture parameters of that mode. In this case, HDG mode remains ON until the airplane arrives at a point where capture can occur. The selected mode then captures and is annunciated in the mode selector and in green letters at the top left side of the PFD/EADI, and HDG cancels. If the autopilot is also engaged, the autopilot receives steering commands according to the selected mode(s). NAV and APR modes can be armed with the HDG mode ON. When intercepting a VOR radial or localizer course with the NAV or APR modes selected, the system switches from ARM to CAP when within the capture limits of the selected VOR radial.

VOR (NAV) and VOR APR (APR)

Two different modes of capture and tracking a VOR signal are used by the Primus 1000 system. One method is used for normal enroute navigation (NAV) and the other for a VOR approach (APR).

For enroute navigation, the desired VOR frequency is selected on a NAV receiver, the course bearing set on the EHSI using the remote instrument controller, and NAV mode is selected on the flight director mode selector. The small green light in the mode selector illuminates, and if the airplane is outside the NAV capture limits, VOR is annunciated in white at the top left of the EADI, and HDG is annunciated in green directly to the right of the white VOR. As the airplane is maneuvered within the capture limits, HDG extinguishes and VOR illuminates in green. When the mode is transitioning to capture, a white box is drawn around VOR for five seconds.

Setup for a VOR approach (APR mode), the desired VOR frequency is selected on the NAV receiver, the course bearing is set on the EHSI, and the APR mode is selected on the flight director mode selector. The green light illuminates in the APR button, and if outside the capture limits, VAPP illuminates in white on the top left side of the EADI and HDG annunciates in green next to VAPP. When the airplane maneuvers into capture range, HDG mode cancels and VAPP annunciates in green in the top left side of the EADI. A white box is drawn around the capturing VAPP for five seconds. The APR mode for VOR approaches (VAPP) increases sensitivity for greater accuracy while conducting VOR approaches.

In both NAV and APR modes, a station passage feature incorporates bank angle limits and a course hold (plus wind drift) mode. The station passage mode for enroute tracking (NAV mode) is of long enough duration to provide a smooth transition of a VOR station at any altitude. The station passage mode for APR mode is of short duration to provide approach accuracy. This does not provide the degree of ride smoothing that is present enroute.

NOTE

VOR approaches without a valid DME signal are prohibited with autopilot coupled or with flight director only.

ILS Approach (LOC or LOC GS)

With a localizer frequency selected in a NAV receiver, operation is similar to capturing and tracking a VOR radial. Selecting APR on the mode control panel with a localizer frequency tuned arms both the LOC and GS modes and engages HDG, if not previously selected and the airplane is outside the capture parameters of the mode. Normally the APR button is pressed when the heading is less than 90° of the final approach course as selected by the remote course knob. HDG is displayed in green at the top left of the EADI, the green light in the APR button of the mode selector illuminates, and LOC and GS are illuminated in white on the upper left and right, respectively, in the EADI. When inside the LOC capture limits, LOC illuminates in green at the top left of the EADI, and HDG extinguishes. At glideslope capture (approximately 1/2 dot), GS il-

luminates in green in the EADI. During transition to both the LOC and GS capture modes, a white box will be drawn around the respective mode annunciations. During ILS approaches, the FD gain is progressively adjusted during the approach using GS deviation, radio altitude, DME, and middle marker passage for gain programming. If the radio altimeter is not operational, this function is performed as a function of glideslope capture and middle marker passage.

The capture limits for VOR and LOC captures are variable depending on DME distance, speed, and intercept angle. Glide-slope capture is locked out until localizer capture occurs. If the localizer mode becomes invalid for any reason, the glide-slope mode is also canceled.

The glide-slope indicator, located on the right side of the EADI presentation, is green unless there is a cross-side selection, in which case it is yellow.

Back-Course Localizer Approach (BC)

Back-course localizer approach capability is provided using either flight director or autopilot or both.

With a localizer frequency set in the selected NAV, selecting BC on the mode selector arms the system for a back-course localizer approach. The front course ILS must be set into the EHSI course pointer to give proper indications on the course deviation bar and for the flight director computer to compute correct back-course corrections during the approach. If back course is set on the EHSI, the command bars and autopilot are given incorrect steering commands. When BC is selected on the mode selector, the green light in the button illuminates and BC is annunciated in white on the left top side of the EADI. HDG may illuminate in green if the airplane is outside of back-course capture parameters. It is imperative to intercept the back course with an intercept angle that is less than 75°. This prevents an inadvertent front course interception. When the back course is captured, the heading annunciator extinguishes and BC is illuminated in green on the top left side of the EADI.

Altitude Hold (ALT) and Altitude Preselect (ASEL)

Selecting altitude hold (ALT) provides steering commands to maintain the altitude at the moment of engagement. An altitude preselect (ASEL) mode provides a preprogramming capability. To use altitude preselect, the desired altitude is set into the ALT window at the lower right corner of the multifunction display (MFD) by means of the knob on the bottom right of the MFD bezel. ASEL illuminates in white in the top right side of the EADI to indicate that the altitude preselect mode is armed. The airplane may be maneuvered toward the desired altitude using any of several methods: the autopilot wheel, touch control steering, FD pitch sync, speed hold, or vertical speed hold. If the airplane is flown manually, the flight director guides the pilot onto the selected altitude. As the airplane approaches the desired altitude, the altitude preselect captures at an altitude corresponding to approximately 1/5 the rate of climb/descent; i.e., at 2,000 feet/minute climb rate, the system captures approximately 400 feet prior to the selected altitude.

At capture, the mode ASEL illuminates in green on the EADI. The flight director performs a smooth level-off at the selected altitude. At level-off altitude, ALT mode is automatically selected and displayed in green on the EADI, and ASEL disappears. Once altitude hold is captured, the touch control steering (TCS) button on the control wheel can be used to change or trim the selected altitude. TCS operates in conjunction with the flight director or the autopilot or both. Once ALT mode is engaged, resetting the BARO setting on the pilot's altimeter causes the airplane to climb or descend to recapture the same indicated altitude. Moving the autopilot pitch wheel causes ALT or ASEL CAP modes to be canceled if either is selected.

Selection of a vertical mode without a lateral mode provides autopilot tracking of the mode, but the FD command bars are not in view.

Vertical Speed Hold (VS)

Vertical speed (VS) hold is selected by pressing the mode button (VS) on the flight director mode control selector. The flight director, autopilot, or both, hold the vertical speed indicated at the moment of engagement. The green light in the mode selector button illuminates and VS illuminates in green on the EADI. Upon initially selecting vertical speed hold mode, the vertical speed synchronizes to the existing vertical speed. Once the vertical speed mode is selected, the pilot can select a different vertical speed with the pitch wheel on the autopilot controller. If the autopilot is engaged after VS mode is selected, the vertical speed must be resynchronized.

The autopilot pitch wheel may be used to change the reference speeds for the vertical speed mode (A/P engaged or disengaged). The touch control steering (TCS) button may also be used to temporarily release the autopilot clutches and maneuver the airplane to a new reference. The vertical speed established when the (TCS) button is released becomes the new reference.

Flight Level Change (FLC)

Activation of the FLC (Flight Level Change) button on the Flight Director Mode Controller selects the FLC mode and overrides all active pitch flight director modes (altitude hold), except V_{NAV}. When V_{NAV} is engaged, activation of the FLC button selects the V_{NAV} sub-mode VFLC.

The IAS/Mach reference is synchronized to the IAS/Mach present at mode activation. Manually selecting a new reference using the pitch wheel on the autopilot controller will cause the system to fly this new reference.

The IAS/Mach speed target comes from the coupled side EADI. Depending on whether the reference is identified as IAS or Mach (based on current altitude), the system will fly the IAS or the Mach reference. Changeover from IAS to Mach (or Mach to IAS) does not cause the reference to move, but simply changes the nature of the digital readout on the EADI; therefore, no aircraft maneuver will occur due to reference change.

The FLC mode is basically an airspeed mode; however, it differs from a standard IAS or Mach mode in the following aspects:

Although the FLC mode, in the long term, tracks the reference airspeed, short-term emphasis is on vertical speed. This minimizes vertical speed excursions due to disturbances or large airspeed changes.

The FLC mode is set up to change flight level, at the selected airspeed, from present altitude to the preselected altitude. It will try to prevent flying away from the preselected altitude target.

GO-Around Mode

Go-around mode (GA) is available through buttons on the left and right throttles. Depressing either button drops all other FD modes on both PFDs and disconnects the autopilot except, for the yaw damper. The FD command bars will command a wings-level 10° nose-up climb attitude. GA illuminates in green on the EADI. After go-around has been selected, the selection of any lateral mode cancels the wings level roll command, but pitch-up command remains. The go-around mode is canceled by selecting another pitch mode, pressing the TCS button, or engaging the autopilot. Depressing a GA button with the A/P engaged in basic mode (F/D not engaged), will not cause the A/P to disengage.

Vertical Navigation (VNAV)

The vertical navigation mode (VNAV) provides a means to define a climb or descent path to a vertical waypoint ahead of the airplane and to track the path to that waypoint. The waypoint is defined based on a distance reference (bias distance) “TO” or “FROM” a short-range VORTAC station waypoint, or the next FMS waypoint if the FMS system is being used for navigation. Upon arrival at the waypoint/ altitude, the mode automatically changes to altitude select (ASEL) capture mode and then to altitude hold (ALT) mode when it levels at the selected altitude.

VNAV DEFINITIONS AND OPERATION

- **Desired Altitude (ALT)** — The altitude at which the airplane levels at the completion of the climb or descent.
- **Station Elevation (STA EL)** — The elevation above sea level of the VORTAC station that the VOR and DME are receiving. Does not apply to FMS waypoints when used for VNAV.
- **TO/FROM Bias (TO/FR)** — The distance set into the VNAV that moves the point for completion of the problem away from the VORTAC or FMS waypoint being used. TO bias moves the point closer to the airplane than the VORTAC or FMS waypoint being used. FROM bias moves the point farther from the airplane than the VORTAC or FMS way point being used.

During VNAV operation, overspeed protection based on the VMO speed limit and underspeed protection based on a fixed 120-knot speed are provided. If either of these speeds is reached, a special sub-mode engages and overrides the VNAV mode until the speed situation is corrected. If a deviation of 1,000 feet from the computed path occurs, VNAV mode cancels.

VNAV operation is canceled if another vertical mode is selected, the air data information from the micro air data computer (MADC) becomes invalid, the DME signal is lost for five seconds, an overspeed or underspeed as described above occurs, the PFD NAV source is changed, glideslope capture or level-off at the waypoint occurs or in case of detection of various system faults by the system monitors.

In order for VNAV mode to operate, the airplane must be proceeding along a direct path toward or away from the short-range NAV (VORTAC) (or to the next FMS NAV waypoint) which has been selected as a reference. If a VORTAC is being used, the VOR azimuth and DME must be locked onto the VORTAC station for VNAV computation. The desired altitude, station elevation (VORTAC only) to the nearest 100 feet, and the TO/FROM bias (if required) must be set in the VNAV system. If the FMS is being used for navigation, the next waypoint may be used, with or without TO or FROM bias, and station elevation (STA EL) data is not required. Attempts to insert VNAV problems behind the airplane or outside the parameters of the system will be ignored by the system.

PROGRAMMING VNAV

Programming is possible when a VOR station is tuned, lock-on of azimuth and DME occurs, and the waypoint desired is within selectable parameters, or when FMS navigation is in use and the next waypoint is used to define the VNAV problem. Arming the VNAV to any waypoint consists of selecting the desired waypoint, and selecting waypoint data which will enable the flight director computer to compute a viable VNAV problem.

VNAV selections can be made using short-range NAV, when a VORTAC station is tuned, identified, and lock-on is achieved. Set the desired altitude in the preselect window. If TO or FROM (FR) bias is required, the second button from the left on the bezel of the multifunction display (MFD) is pressed which results in display of a box to program TO or FR bias by turning the left knob on the MFD. TO or FROM is selected before the distance selection is made by toggling the button, resulting in annunciation of TO or FR above the selection window. Station elevation (STA EL) of the VORTAC station in use is then set by pressing the second button from the right and setting the correct elevation, to the nearest 100 feet, into the window above it. The VNAV problem is now established, and VNAV may be selected.

If long-range NAV is used, the problem is similarly defined. FMS must be selected on the display controller, which results in long range data being displayed on the menu at the bottom of the MFD display. Program VNAV as discussed prior. Station elevation (STA EL) is not required.

If a valid problem has been defined, the computed angle will be displayed on the MFD VNAV menu located at the bottom right of the MFD display. A NAV problem is valid only if the vertical angle is less than ±6°. The flight director computer will continually compute the vertical angle based on aircraft position and update the display on the vertical path indicator on the PFD. If the pilot desires, he can rotate the VNAV set knob and increase the vertical angle up to a maximum of 6°, which creates a vertical path intercept point some distance ahead of the aircraft. Once a valid VNAV problem has been defined, the pilot can select the VNAV mode on the FD mode selector. VNAV mode will not activate until it is selected, or selection is affirmed, by pilot action. Adjacent to the calculated VANG display is a vertical speed (VS) display. It is used for monitoring the climb or descent and cannot be set.

If the pilot has selected an intercept point ahead of the airplane by increasing the vertical angle before selecting the VNAV mode, the flight director remains in the previous mode until the appropriate time. Approximately one minute prior to the flare point the altitude alert horn sounds two short beeps. The vertical track alert (VTA) on the PFD and the VNAV annunciator on the FD mode selector flash. Pilot action is required before the VNAV capture phase can commence. The pilot must press the flashing VNAV button on the mode selector before it stops flashing to allow the mode to capture. Once the button is pressed, annunciation in the mode selector stops flashing and remains on, and the flashing VTA annunciator on the PFD becomes steady . If the pilot wishes to cancel the mode, he can press the VNAV button twice on the mode selector when it flashes, or he can do nothing and wait for the flashing to stop, at which time the mode automatically disengages.

When the VNAV mode is engaged, the VNAV parameters are frozen. This includes STA EL, TO, FROM, and VANG; changing the ALT SEL value also causes the mode to drop out. The pilot may still view any of these parameters, but the set knob will have no affect. After the airplane has leveled off at the waypoint altitude and transitioned into altitude hold mode, the VNAV parameters for the current problem are erased.

If the pilot deselects the VNAV mode by pressing the VNAV button, the flight director cancels the mode, but data for the current waypoint are retained. The angle from the present position to the waypoint is still tracked, but the parameters are no longer frozen and can be modified as desired by the pilot. The VNAV mode can be reselected as long as the problem remains valid.

Altitude Alert

The altitude alert system is automatically engaged in conjunction with the altitude preselect mode (ASEL) and the vertical navigation (VNAV) mode. The desired altitude is set into the system for use of VNAV or ASEL modes. In both cases the altitude is set into the lower right comer of the MFD with the right knob on the MFD bezel. The desired flight director mode which is to be used to reach the designated altitude is then selected on the flight director/autopilot mode control panel. Refer to Altitude Hold and Altitude Preselect, above. If the pilot does not desire to select a flight director mode, the airplane may be flown manually, and the altitude alerting system will still provide the appropriate annunciations.

NOTE

Airplanes 5001 thru 5155 not incorporating SB560XL-24-14. Single-point VNAV will remain armed if another vertical mode is selected for early descent prior to path capture. After level-off, a descent away from the altitude preselector will occur upon intercept of the programmed VNAV path. To prevent an undesired descent after level-off, manually disarm the VNAV mode.

MODE ANNUNCIATIONS

Flight director vertical and lateral modes are annunciated along the top of the PFDs. Armed modes are annunciated in white slightly to the left of the captured vertical and lateral mode annunciations, which are displayed in green. Lateral modes are displayed to the left of top center and vertical modes displayed to the right of top center on the PFDs. A white box appears around a capture or hold mode for five seconds after mode transition from armed to capture. A summary of the lateral and vertical mode annunciations and transitions are listed below:

- **VOR —** A NAV mode (VOR) is armed or has been captured and is being tracked.
- **HDG —** Heading select mode is engaged.
- **LOC —** Localizer has been armed or captured.
- **VAPP —** VOR approach is selected, or course captured has occurred.
- **GS —** Glide slope is armed or captured.
- **ASEL —** Altitude preselect is armed (white); altitude preselect transition (green).
- **ALT —** Altitude hold mode is engaged.
- **BC —** Back course is armed or captured.
- **VS —** Vertical speed hold has been selected and captured.
- **FLC —** Flight Level Change Mode has been selected and captured.
- **V-NAV —** V-VNAV mode is armed or captured.
- **LNAV —** Long-range NAV (FMS) mode has been selected.
- **GA —** Go-around mode has been selected.

MISCELLANEOUS ANNUNCIATIONS

ATT1 (or ATT2) — Attitude source (amber for “cross-selection”).

DH — DH box in view on the PFD, left of the ADI, is in view 100 feet above the selected DH height. DH illuminates amber in the box when the airplane reaches the preset decision height.

Lateral Transitions:

- VOR arm to VOR cap
- LOC arm to LOC cap
- BC arm to BC cap
- VAPP arm to VAPP cap

Vertical Transitions:

- VNAV arm to VNAV cap
- VNAV cap to ALT
- ASEL arm to ASEL cap
- ASEL cap to ALT hold
- GS arm to GS cap

AP ENG — AUTOPILOT ENGAGED (green). A green arrow points either left or right, indicating to which flight director (pilot’s or copilot’s) the autopilot is coupled for guidance.

TCS ENG — Illuminates in amber to indicate touch control steering is engaged.

AP TEST — Illuminates in amber when the autopilot is in test mode. Annunciation is automatic immediately after power-up. It is normally not in view due to warm up time for PFDs.

TRN KNB — Illuminates in amber when the autopilot turn knob is out of the center detent.

COMPARISON MONITOR

Annunciations

Selected pilot and copilot input data are compared in the symbol generator. If the difference between the data exceeds predetermined levels, an out-of-tolerance symbol is displayed on the PFD in amber. A list of the compared signals and the displayed cautionary symbols is given in Table 16-2. When the compared pitch and roll attitude or glide-slope and localizer signals are out of tolerance, a combined level (ATT or ILS) is displayed.

EFIS Equipment Failure Summary

Display Reversion

In the event of failure of one of the PFDs, turning off the failed display DIM knob of the respective display controller causes that display to be presented on the multifunction display (MFD). Control of the PFD is still through the respective display controller.

EFIS Equipment Failure Checklist

Failure of equipment providing information to the EFIS is annunciated by flags or dashes. Failure effects of EFIS equipment are listed in Table 16-3.

Full counterclockwise OFF position of the DIM knob turns off the failed display and selects the respective display to the multifunction display (MFD) tube.

For detailed information concerning operations of the Primus 1000 system, consult the Honeywell P-1000 Integrated Avionics System Pilot's Manual for the Citation EXCEL.

Table 16-2 COMPARISON MONITOR ANNUNCIATIONS

COMPARED PARAMETER	ANNUNCIATION	TRIGGERING DIFFERENCE
PITCH ATTITUDE	PIT	5°
ROLL ATTITUDE	ROL	6°
HEADING	HDG*	6°
LOCALIZER	LOC**	APPROX. 1/2 DOT
GLIDE SLOPE	GS**	APPROX. 1/2 DOT
PITCH AND ROLL ATTITUDE	ATT	5° AND 6°, RESPECTIVELY
LOCALIZER AND GLIDESLOPE	ILS**	1/2 AND 1/2 DOT, RESPECTIVELY
INDICATED AIRSPEED	IAS***	5 KNOTS
ALTITUDE	ALT***	200 FEET

* IF THE COMPARED HEADING SOURCES ARE NOT THE SAME (BOTH MAG OR TRU), THE COMPARISON MONITOR IS DISABLED.

** THESE COMPARISONS ARE ACTIVE ONLY DURING FLIGHT DIRECTOR, LOCALIZER, AND GLIDE-SLOPE CAPTURE WITH BOTH NAV RECEIVERS TUNED TO THE SAME LOC FREQUENCY.

*** AIRSPEED AND ALTITUDE DISPLAYS FLASH FOR TEN SECONDS AND THEN GO STEADY.

EMERGENCY FLIGHT INSTRUMENTS

STANDBY FLIGHT DISPLAY SYSTEM (MEGGITT)

The MEGGITT tube is a DC-powered cathode ray tube indicator combining standby attitude indicator, altimeter, and airspeed indications into one composite instrument (Figure 16-12). A Mach indication is also included in the instrument.

The Standby Flight Display (SFD) contains solid state inertial sensors for the measurement and presentation of aircraft pitch and bank attitudes. Application of 28-volt DC power to the display system initiates the attitude initialization process, which is identified by the display of the message "attitude initializing" in yellow on the SFD. The duration of the initialization process is normally 180 seconds. The aircraft should not be moved until the SFD is initialized (timer extinguishes on the face of the instrument).

The attitude display has an instantaneous display range of 360° of bank and 50° of pitch. A moving tape on the right side of the display includes a "rolling digit" depiction of altitude; the tape is calibrated in 100 foot increments. Baro data is set in the altitude display by a knob on the bottom right of the bezel; clockwise rotation increases the pressure setting and counterclockwise decreases it. The setting is displayed simultaneously in millibars at the top right of the display and in inches of mercury at the bottom right. On the left side of the display is a moving tape showing airspeed. The tape is marked in ten knot increments with a "rolling digit" display in the center. The airspeed display becomes active at 40 knots. The Mach number is displayed in the upper left corner of the display. The Mach display range is 0.35 to 0.999 Mach.

Failure flag indications for airspeed and altitude are red crosses covering the appropriate tape box, with all indications removed from within the box. The failure flags for the Mach indication and Baro Setting are a series of four red dashes in the appropriate display area.

Table 16-3 EFIS EQUIPMENT FAILURE CHECKLIST

FAILURE	ANNUNCIATION	FLIGHT DIRECTOR	PILOT ACTION
Symbol Generator Failure	Red X on PFD or Display Blank	All modes cancelled	Select opposite SG on MFD
Display Controller Failure	Display Cannot be Changed	N/A	Select opposite SG on MFD display controller
PFD Failure	Display Goes Blank	None	Revert display to the MFD display
Heading Failure	Red HDG FAIL on EHSI, bearing pointers, etc., removed	Command Bars out of view	Select opposite AHRS map, heading source by pressing the appropriate HDG REV button.
Attitude Failure	ATT FAIL annunciation: no pitch or scale or roll pointer, sphere all blue	None	Select opposite AHRS attitude source by pressing the appropriate ATT REV button
Course Deviation Failure	Red X through scale and course deviation pointer removed	Command bars, CDI pointer, and applicable pointer off	Revert display to the MFD display
Flight Director Failure	FD FAIL on PFD	FD cues and mode	Select opposite flight director on AP/FD PFD1 - AP/FD PFD2 switch.

A light sensor is located on the bottom left side of the instrument case. It provides ambient light level data to the backlight control system to ensure optimum display brightness. The lighting level can still be controlled manually from the center instrument panel light rheostat control.

The navigation display is selected by the APR button on the bottom of the display bezel. Pressing the button once will display ILS localizer and glideslope flight director information on the Meggitt tube, provided the NAV 1 receiver is tuned to an ILS. Pressing the button a second time will display Back Course localizer information on the Meggitt tube, provided the NAV1 receiver is tuned to a localizer back course frequency. Pressing the button a third time will remove all navigation information from the Meggitt tube. VOR tracking is not available. The standby HSI will display all navigation information (ILS, BC, VOR) from the NAV1 receiver.

Power to the standby flight display is controlled by a switch marked STBY PWR - ON/OFF/TEST located on the pilot's lower instrument panel. The SFD has an emergency source of power from an emergency battery pack located in the nose avionics compartment. If bus voltage falls below a minimum amount, the standby power relay will activate and SFD power will be supplied from the battery pack. This battery pack also provides emergency instrument lighting for the standby flight display, the dual fan (N_1) tachometers, and the standby Horizontal Situation Indicator (HSI).

The battery pack is constantly charged by the airplane's electrical system, and should be fully charged in the event of an electrical power failure. The standby instrument power switch must be ON for automatic transfer to battery power to occur. The SFD will operate for a minimum of 30 minutes on emergency battery pack power. An amber STBY PWR ON light next to the STBY PWR switch illuminates when the SFD is turned ON and the airplane's electrical system is not charging the SFD emergency battery pack (emergency battery pack is powering the SFD). When the SFD switch is held to the spring-loaded TEST position, a self-test of the battery and circuits is accomplished. The green STBY PWR TEST light next to the STBY PWR switch will illuminate if the test is satisfactory and the battery is sufficiently charged.

Maximum allowable airspeed (V_{MO}) is displayed in analog form by a red warning strip on the airspeed tape. When V_{MO} is reached, the numerals on the numeric airspeed display change from white to red. When the maximum allowable Mach number (M_{MO}) is reached, the numeric Mach number display will also change from white to red.

A built-in test system (BIT) will automatically detect any failure of the display at power up or during continuous operation. If a failure is detected, the appropriate part of the display is replaced with a message indicating the failure. Where it is not possible to display an appropriate message, the display back-lighting is switched off.

STANDBY HORIZONTAL SITUATION INDICATOR (HSI)

The standby horizontal situation indicator is a three-inch instrument located on the pilot's instrument panel (Figure 16-12). It provides navigational guidance in case of PFD/flight director failure, and is powered by the Emergency Bus.

The standby HSI displays compass heading, (No. 2 AHRS) glide-slope, and localizer deviation and airplane position relative to VOR radials. The compass card is graduated in 5° increments, and a lubber line is fixed at the fore and aft positions. A fixed reference airplane is in the center of the HSI, aligned longitudinally with the lubber line markings.

The course cursor is set by a knob on the instrument. Once set, the cursor rotates in its set position with the compass card. The course deviation bar, which forms the inner segment of the course cursor, rotates with the course cursor.

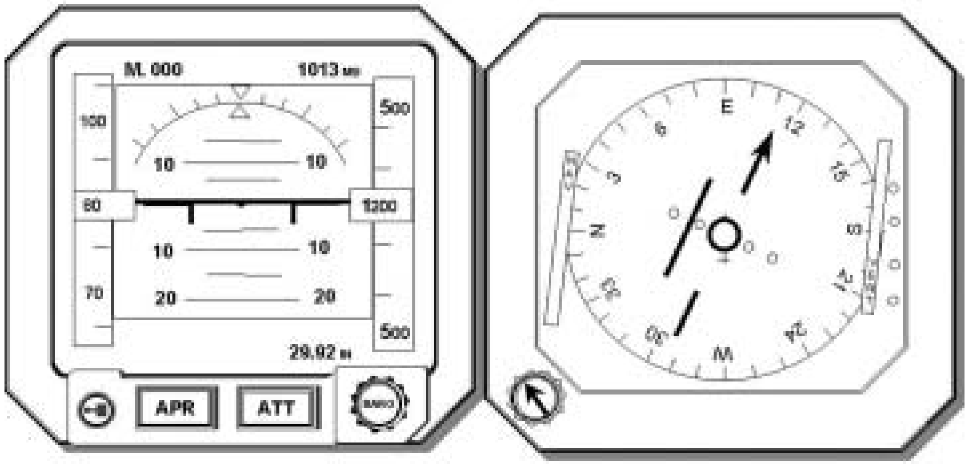


Figure 16-12 Meggitt Tube and Standby HSI

A blue needle, which displays ADF1 bearings, rotates around the outer portion of the dial.

A heading (HDG) flag appears in the instrument when the compass system is OFF. If the heading signal from the No. 2 AHRS becomes invalid, primary power to the indicator is lost, or the error between the displayed heading and the received signal becomes excessive the HDG flag will appear.

The course deviation bar moves laterally in relation to the course cursor. Course deviation dots in the HSI act as a displacement reference for the course deviation bar. When tracking a VOR, the outer dot represents 10°, while on an ILS localizer it represents 2 1/2° White TO-FROM flags point to or from a station along a VOR radial.

A red warning flag comes into view when power is OFF, NAV information is unreliable, or signals from the NAV receiver are not valid. The standby HSI displays only NAV1 information.

The glide-slope deviation pointer is located to the right side of the display. When receiving glide-slope information during an ILS approach, the green deviation pointer is uncovered by the red VERT warning flag, which otherwise is in evidence. If an ILS frequency is not tuned and being received, or the ILS signal is unusable or unreliable, the deviation pointer is covered by the red warning flag.

MISCELLANEOUS FLIGHT INSTRUMENTS

RAM-AIR TEMPERATURE (RAT) INDICATOR

A digital ram-air temperature (RAT) indicator, located on the upper left side of the center instrument panel, displays air temperature uncorrected for ram rise. Either Celsius or Fahrenheit may be selected by a switch on the indicator. Temperature sensing is taken from the RH engine EEC temp sensor (T.O.), located in the right engine inlet. If the right engine T.O. sensor fails, the #2 MADC will automatically provide temperature information to the RAT indicator.

MAGNETIC COMPASS

A standard liquid-filled magnetic compass is mounted above the glareshield. Directly above the compass are the seating height indicator balls.

FLIGHT HOUR METER

The flight hour meter, located on the copilot's upper instrument panel, displays the total flight time on the airplane in hours and tenths. The left landing gear squat switch activates the meter when airplane weight is off the gear. A small indicator on the face of the instrument rotates when the hour meter is in operation.

DIGITAL CLOCK (DAVTRON)

Two Davtron model M877 clocks, located on the pilot's and copilot's upper instrument panels, can display four functions: local time, GMT, flight time, and elapsed time. Two versions of the elapsed time function may be selected: count up or count down.

The clock has two control buttons: SEL (select) and CTL (control). The SEL button is used to select the desired function, and the CTL button to start and reset the selected mode.

For normal operation, either local time or Greenwich Mean Time (GMT) may be selected. GMT is displayed only in 24-hour format, and local time is 12-hour format. Pressing the SEL button sequentially displays GMT, local time, flight time, and elapsed time. The display mode is annunciated GMT, LT, FT, and ET, as applicable, under the time display window.

To set GMT or local time, select the desired function by pressing the SEL button. Simultaneously press both the SEL and the CTL buttons to enter the set mode. The tens of hours digit will start flashing and may be incremented by pressing the CTL button. The next digit is then selected by pressing the SEL button, and similarly set by means of the CTL button. When the last digit has been set, press the SEL button to exit the set mode. At that time the clock starts running and the illuminated annunciator resumes flashing.

The clock may be used as a stop watch to time approaches, etc. Select ET with the SEL button, and press the CTL button to start the timing. The clock starts counting elapsed time in minutes and seconds up to 59 minutes and 59 seconds. It then switches to hours and minutes and continues up to 99 hours and 59 minutes. Pressing the CTL button resets the elapsed time to zero.

To use the clock for an elapsed time "countdown" display, select ET for display, and enter set mode by pressing both buttons simultaneously. A maximum countdown time of 59 minutes and 59 seconds can be set. The time from which it is desired to count is entered in the same manner as setting GMT or local time. When the last digit is set, press the SEL button to exit the set mode. Pressing the CTL button starts the countdown. The display flashes when the time reaches zero. After reaching zero, the ET counter counts up. Pressing the CTL button again resets ET to zero.

Flight time mode is enabled by a landing gear squat switch, which causes the clock to operate any time the airplane weight is off the landing gear. Flight time may be reset to zero by selecting FT mode with the SEL button and holding down the CTL button for three seconds. Flight time is zeroed when the CTL button is released. A total of 99 hours and 59 minutes can be shown.

A flight time alarm mode flashes the clock display when the desired flight time is reached. To set the alarm function, select FT with the SEL button, and enter the set mode by pressing both buttons simultaneously. Enter the desired alarm time in the identical manner that GMT or local time is set. When flight time equals the alarm time, the display flashes. If FT is not being displayed when the alarm time is reached, the clock automatically selects FT for display. Pressing either the SEL or CTL button turns off the alarm and resets the alarm time to zero. Flight time is unchanged and continues counting.

The clock display may be tested when power is on the airplane by holding the SEL button down for three seconds. The display shows 88:88, and all four annunciators are activated.

STALL WARNING AND ANGLE-OF-ATTACK SYSTEM

The angle-of-attack system is powered by 28V DC from the left main DC bus and incorporates an angle-of-airflow sensor, a signal summing unit, a vane heater monitor, an angle-of-attack indicator, a stick shaker, and an optional indexer.

The vane-type angle-of-airflow sensor, which is located on the forward right side of the fuselage, detects the angle of airflow and deflects accordingly. The wedge-shaped vane streamlines with the relative airflow and causes a transducer to send signals to the signal summing unit (computer). Signal inputs concerning flap position are also received by the signal summing unit. It then compensates for that variable and transmits the information to the angle-of-attack indicator and the indexer. Indications are accurate throughout the weight and CG range of the airplane.

The full-range-type indicator is calibrated from 0.1 to 1.0, and marked with red, yellow, and white arcs. Lift information is displayed on the indicator with 0.1 representing near zero lift and 1.0 representing stall. Lift being produced is displayed as a percentage and, with flap position information, is valid for all airplane configurations and weights. At 1.0 where full stall occurs, 100% of the available lift coefficient is being achieved. At the bottom of the scale (0.1) near zero lift is being produced. The area at the lower part of the scale (0.57 to 0.1) represents the normal operating range, except for approach and landing. The narrow white arc (0.57 to 0.63) covers the approach and landing range, and the middle of the white arc (0.6) represents the optimum landing approach (V_{APP} or V_{REF}). The yellow range (0.63 to 0.85) represents a caution area where the airplane is approaching a critical angle of attack. The red arc (0.85 to 1.0) is a warning zone that represents the area just prior to stick shaker activation and continuing to full stall. At an indication of approximately 0.79 to 0.88 (depending on flap setting and rate of deceleration) in the warning range, the stick shaker activates.

If the angle-of-attack system loses power or becomes inoperative for other reasons, the needle deflects to the top of the scale and stows at a 1.0 indication.

NOTE

The airplane must not be flown if the stick shaker is found to be inoperative on the preflight check or if the angle-of-attack system is otherwise inoperative.

Stick shakers are located on the pilot's and copilot's control columns about 9 inches down from the control wheels and on the forward side. The stick shaker provides tactile warning of impending stall. The angle-of-attack transmitter causes the stick shaker to be powered when the proper threshold is reached.

WARNING

IF THE ANGLE-OF-ATTACK VANE HEATER FAILS AND THE VANE BECOMES ICED, THE STICK SHAKER MAY NOT OPERATE OR MAY ACTIVATE AT NORMAL APPROACH SPEEDS.

The approach indexer, mounted on the pilot's glareshield, provides a "heads-up" display of deviation from the approach reference. The display is in the form of three illuminated symbols which are used to indicate the airplane angle of attack. High angle of attack is analogous to low airspeed; low angle of attack is analogous to high airspeed.

Illumination of the symbol is progressive as the airplane angle of attack changes. When the airplane speed is on reference, the green center circle is illuminated. As the speed decreases from reference (0.6), the circle illumination dims and the top red chevron illumination increases until the top chevron is

fully illuminated and the circle is extinguished. As the angle of attack becomes high, the top red chevron begins to flash.

When the airplane is accelerating from the on-speed reference, the illumination of the green circle dims and illumination of the bottom yellow chevron increases until the circle is extinguished and only the bottom chevron is illuminated.

The top red chevron points down, indicating that the angle of attack must be decreased to eliminate the deviation. The bottom yellow chevron points up to indicate that the angle of attack must be increased to eliminate the deviation.

The indexer is active any time the nose gear is down and locked and the airplane is not on the ground. There is a 20-second delay after takeoff before the indexer activates.

Stall strips on the leading edge of each wing create turbulent airflow at high angles of attack, causing a buffet to warn of approaching stall conditions. They are a backup to the angle-of-attack stick shaker system in case of malfunctions and electrical power failures.

COMMUNICATION/ NAVIGATION

HONEYWELL PRIMUS II REMOTE RADIO SYSTEM (RMU)

VHF COMM (Note: The current RMUs are capable of 8.33khz spacing)

The RCZ-850 integrated communications unit normally operates in the frequency range of 118.00 to 136.97 MHz. The unit can be strapped to extend the upper range to 152 MHz for operation in parts of the world where those frequencies are used. The RCZ-850 unit is the communications component of the SRZ-850 integrated radio system. The COM radios are controlled from the RM-850 radio management unit (RMU), two of which are mounted on the center instrument panel (Figure 16-13). COM 1, NAV 1, ADF 1, etc., are controlled by the left RMU. The COM 2, NAV2, and ADF 2 (if installed) are controlled by the right RMU. The unit being controlled is annunciated on the control display unit of the RMU. The four radio functions: COM, NAV, ATC (transponder), and ADF which are controlled by the RMU are all displayed on page one (main fre-

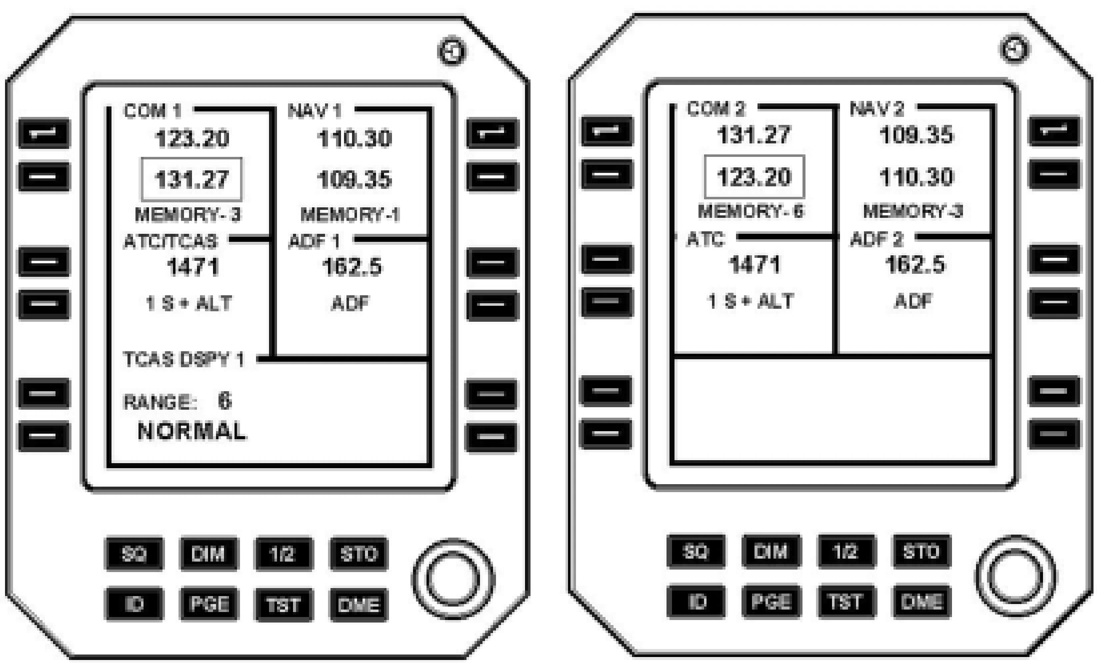


Figure 16-13 Radio Management Units (RMU)

quency select page) of the RMU. Tuning control for the desired function/parameter is obtained by pressing the line select key next to that function/parameter. The COM radio has a memory capacity for up to 12 frequencies to be selected and stored for later use.

Controls and Indicators

Normally the COM radios are controlled through the controls and display located in the upper left corner of the radio management unit (RMU). Any selectable parameter is changed by pressing the corresponding line key next to the displayed parameter. This brings an amber box (cursor) to surround that position, which allows it to be tuned by the single controller tuning knob on the bottom of the RMU.

Tuning of the COM radios is accomplished by three methods. The first method, discussed below, also provides methods to store frequencies in the memory locations. This is considered the "normal" method. Storing of the frequencies while tuning is not required; however, it is discussed here only because it may be convenient to store the frequencies as they are used for later use. The second method is "direct tuning," and the third method is remote tuning through the auxiliary Standby Radio Control Unit (SRC). This may be used when only battery power is available or desired, or in case of an emergency. Operation of the STH control head is discussed later in this section.

Normal, or preselect tuning of the COM radios is accomplished in the following manner: press the line key next to the second COM frequency line displayed on the RMU-the amber box will move to that position if it is not already there; set the desired frequency by means of the concentric tuning knobs at the bottom of the RMU; press the upper left button on the RMU bezel (the one with vertical arrows), which will switch the pretuned frequency with the active frequency. When a frequency is preselected (set in the second line), this may result in the changing of a frequency which was identified by MEMORY, plus a number from 1 to 12, below the active frequency. The prior number has been stored in memory and the imposition of the second frequency over it is only temporary (which is identified TEMP). This will not result in the new frequency being stored in the memory unless the STO button is pressed before the frequency is transferred to the active location (top line). In this case, the word TEMP will be replaced by the word MEMORY plus the memory position number. The pilot may progress through all 12 of the memory locations by pressing the line key near the line identified by TEMP or MEMORY in the COM box (upper left hand corner), which will move the amber box to surround that line. Turning either the large or small tuning knob will then select each memory space sequentially, showing the frequency stored there in blue on the line above the MEMORY annunciator line. Vacant memory locations will not appear. When the last occupied memory location is selected, the frequency shown on the second line, which was a temporary frequency in memory, will again be shown to occupy that space, plus the word TEMP, indicating that it is not stored in MEMORY.

When progressing through the stored memory locations, the frequency in the memory location being displayed can be transferred into the active position (tuned) simply by pressing the upper button (the one with the vertical arrows).

If the pilot desires to view all of the stored frequencies at once, he may press the PGE (page) button at the bottom of the RMU and the active frequency, with a maximum of six stored frequencies, will be displayed along with the number of their memory location. Pressing the line key adjacent to the MORE annunciator will advance the page to show the remaining frequencies with their location numbers of 7 through 12. If it is desired to insert a frequency in any particular location on these pages, move the cursor to that location by pressing the line key next to the desired memory location and the tuning knob will control that selection. The memory locations must be filled sequentially, i.e., blanks cannot be left open. If memory location eleven is vacant, for instance, and an attempt is made to store a frequency in location twelve, the word "can't"

will appear in amber at the bottom of the page. It is not necessary to push STO to store the frequency. If deletion of a stored frequency is desired, press the line key adjacent to that memory location and press the line key adjacent to the DELETE annunciator. Higher memory locations will move down to fill the vacant space. If the pilot desires to place a frequency in a particular memory location, press the line key at that location to move the amber box there; press the line key at the INSERT location. The frequencies at the selected location and at higher location numbers will move up one location.

The frequency in the selected location may then be modified and it will be stored.

If all the memory locations on the first memory page are not filled, the second memory page cannot be accessed.

Direct tuning of the COM radio is accomplished by selecting the cursor (amber box) to the COM preset location (second frequency line), and pressing the line key at that position for a minimum of three seconds. The preset frequency will disappear and the cursor will move and enclose the active frequency. Direct tuning is then available. Preset tuning may be restored by pressing the same button again.

An additional feature provided by the SRZ-850 integrated system, is stuck microphone protection. The COM transmitter has a two-minute timer which cuts off transmission after that time has elapsed if the MIC key has not been released. A short warning tone is sounded a few seconds before the automatic shutoff. When the microphone cutoff has been activated at the two-minute limit, a MIC STK warning in red will be annunciated in the upper left corner of the RMU.

A TX annunciation at the top of the COM frequency window will annunciate whenever the transmitter is active.

When the second (first memory location) page of the display is selected, a “NARROW BANDWIDTH SELECT” annunciation will appear in the upper right corner of the display. Narrow bandwidth is the normal selection; however, a wider bandwidth may be selected for use in areas where slightly off-channel transmitters are used. Its selection will result in improved reception in such areas. The selection is made by pressing the double arrow selector next to the annunciation. Another press of the selector will return the selection to the original.

If any of the components of the radio system fail to respond to tuning or operating commands of the RMU, the frequency or operating command associated with that particular function will be dashed out. This alerts the crew to a failure or abnormal system operation.

“Cross-side” operation of the RMU is possible by pressing the 1/2 button on the bottom of the RMU. This allows the operator to tune the opposite side radio system from that RMU. The tuning will be followed on the other RMU and so indicated. The system banners will be indicated in magenta color to serve as a reminder of the cross-tuning condition.

Each time the integrated radio system is powered up with the landing gear squat switches activated, a power on self-test (POST) will be activated. If any radio or bus fails any test parameter, an error message will be displayed on a test results page. If no errors are detected, the main tuning page will be displayed.

A pilot activated self-test (PAST) may be initiated by pressing the TST button on the RMU. A complete test will then be accomplished on the component represented by the window at which the yellow cursor is located. At the completion of the test, a legend will appear in the window for a short time to indicate successful completion. If the test is not successful, an error message will appear to indicate which circuit area has failed.

By pressing the DIM button on the bottom of the RMU, the tuning button may be used to dim the display. Exit from the dim mode is accomplished by pressing the DIM button again. Variations in ambient light will be automatically sensed, within limits, and automatically adjusted to maintain a desired setting.

NOTE

RMU 1 is powered from the emergency DC bus and RMU 2 is powered from main DC power.

VHF-NAV

The RNZ-850 integrated navigation unit operates in the frequency range of 108.00 to 117.95 MHz. The RNZ-850 system encompasses the functions of VHF NAV, localizer and glideslope receiver, and marker beacon receiver, as well as the addition of the ADF and DME functions which, in conventional systems, are separate units. Operation of the ADF and DME modes will be covered in the section where operation of the standard ADF and DME installations are discussed. Operation of the marker beacon system is discussed under "Marker Beacon" below.

Glideslope paired frequencies are tuned with the published ILS frequencies as in standard VHF NAV practice. The RNZ-850 is the navigation component of the SRZ-850 integrated radio system. The two NAV integrated receivers are controlled and tuned in a similar manner to the RCA-850 COM units discussed under "Primus II Remote Radio System-COM." A minor difference is the requirement for the PGE (page) button to be pressed twice in order to access the NAV page which shows the first six NAV memory locations. Otherwise, changing, storing and deleting frequencies is accomplished in the same manner.

The NAV frequency window on the main tuning (first) page has an additional function called the "DME Split Tuning Mode." This function involves "DME hold" plus some additional features, and is discussed under "Distance Measuring Equipment" in the Pulse Equipment part of this section.

NAV1 is also tunable by the Standby Tuning Head. Tuning of the STH is discussed under "Standby Tuning Head" in this section.

Both NAV1 and NAV2 are selectable on the pilot's and/or copilot's DC-550 display controller to be displayed on the respective PFD. If both PFDs are displaying the same NAV source, the annunciation will be in amber.

Operation of the NAV displays on the standby horizontal situation indicator (HSI) is discussed in the NAV section.

ADF — NAV

The automatic direction finder (ADF) function of the Primus II Remote Radio System is provided by the DF-850 ADF receiver module, which is a component of the RNZ-850 integrated navigation unit. As discussed in the COM section above, the tuning of the complete system, which includes the ADF, is accomplished by means of the remote management unit (RMU), the RM-850.

The receiver has a frequency range of 100.00 to 1799.5 kHz in 0.5 increments. A strap selectable option is available which allows tuning of marine emergency frequency of 2181 through 2183 kHz.

Four modes of operation are available on the DF-850 ADF: ANT (Antenna), ADF (Automatic Direction Finder), BFO (Beat Frequency Oscillator), and VOICE. In ANT mode, the ADF receives only and does not compute bearing information. In ADF mode, the system receives signals and computes relative bearing to station. In BFO mode, a beat frequency oscillator is added to the signal for reception of CW signals. In VOICE mode, the reception bandwidth is widened for improved voice audio on the frequency. The VOICE mode is not used for navigation. Bearing information is available only in ADF and BFO modes. If ANT is used for tuning, random ADF needle searching is prevented. The modes are selected by pressing the lower line key adjacent to the ADF window. Progression is: ANT; ADF; BFO; and VOICE. The mode changes each time the line key is pressed. When the tuning cursor (amber box) surrounds the lower ADF line, the ANT, ADF, BFO, and VOICE progression may also be selected by turning the tuning knob.

When the line select key adjacent to the frequency window of the ADF is pressed, the cursor will move to the ADF frequency window and the ADF may be tuned by the tuning knobs. Tuning is in 0.5 kHz increments with the small knob and 10 kHz with the large knob. If the knobs are turned faster larger increments are selected for each turn enabling large changes to be made in much less time. The rate of increased tuning speed is proportional to the rate the knobs are turned.

The ADF has a "scratch pad" memory which will store one frequency. This is accomplished by selecting the desired frequency and pressing the STO button for two seconds. To retrieve the frequency from memory, press the line select key adjacent to the ADF frequency window for two seconds.

The ADF bearing information may be selected on either the "O" or "diamond" bearing needles of either pilot's electronic primary flight displays (PFDs) in single ADF installations. If dual ADFs are installed, the "O" bearing pointer will display ADF 1, and the "diamond" bearing pointer will display ADF 2, when selected. Selection is accomplished by means of the bearing knobs ("O" and/or "diamond") on the respective DC-550.

ATC TRANSPONDER

The ATC (transponder) function of the optional SRZ-850 integrated radio system is provided by the XS-850 transponder module, which is a sub-unit of the RCZ-850 integrated communication unit. It functions as a 4096 code mode A transponder, as well as providing mode C (altitude) and mode S (collision avoidance) information. Altitude information is provided by the respective (1 or 2) AZ-850 micro air data computer in the pilot's or copilot's Primus 1000 system.

General tuning information concerning the SRZ-850 system is discussed under "Primus II Remote Radio System-COM" in this section. Specifically, tuning of the transponder is accomplished by pressing the line key adjacent to the desired ATC function on the left side of the main tuning page which is displayed on the RMU. The ATC window has two lines. The top line represents the tune able transponder codes and the second line represents the transponder modes. When the line key adjacent to the transponder code line is pressed, the amber box (cursor) will surround the code digits, which are then tunable by the tuning knobs. The large knob controls the left two digits, and the small knob controls the right two digits.

Pressing the mode select line button moves the cursor box to the mode select annunciator which connects the tuning knobs to that window. Either knob may then be used to select modes in the following sequence:

ON — System in Modes S and A, no altitude is reported.

ON+ALT — Replies on Modes A, C and S.

S ONLY — Mode S operation only.

S+ALT — Mode S with altitude reporting (Mode C).

Only one transponder is in operation at one time; the opposite one is held in STANDBY for instantaneous operation, if required. The system in operation is controlled by the mode select line key. Pressing the mode select line key (once the cursor is moved to that line) cycles the transponders as follows:

STANDBY — Both units in STANDBY.

SYSTEM No. 1 in operation.

STANDBY—Both units in STANDBY.

SYSTEM No. 2 in operation.

STANDBY—(sequence will repeat).

The system in operation is indicated by a "1" or "2" in front of the selected mode.

A transponder code may be stored in memory. To accomplish this, select the desired codes and press the STO button for two seconds. To retrieve the code from memory, press the line select button for two seconds. The IDENT function of the transponder may be activated by pressing the ID button on the RMU or by pressing the ID button on the inboard side of either the pilot's or copilot's control wheel. Pressing any ID button will activate the ID mode for approximately 18 seconds. An amber ID annunciation will appear along the top edge of the transponder window during ID mode activation.

DME NAV

The optional Primus II DME system is comprised of systems which are organized into compact modules. Each module, concerning the DME system, is comprised of an RNZ-850 integrated navigation unit, an NV-850 VHF NAV receiver and a DME-850 distance measuring module. The DME transmitter of the DME-850 works in the L frequency band, and the receiver frequency range is from 962 to 1213 MHz. DME tuning normally follows the VHF NAV receiver tuning which selects the DME frequencies paired to the VHF VORTAC published frequencies. The Primus II, however, has a special "hold" function which also allows the tuning of military TACAN channels in order to receive the DME portion of the TACAN signals.

The DME has the capability to scan six channels, simultaneously tracking four selected DME channels for distance, ground speed and time-to-station, as well as tracking two stations for identification (IDENT) functions. Of the four channels which can track three functions (DIST, GS and TTG), two are dedicated to the flight management system (FMS).

Normally, one DME station will be tuned to an active VOR frequency, which is annunciated on the top line of the NAV tuning window of the radio management unit (RMU). Another (preset) VOR frequency may be selected in the preset frequency window. When a frequency is set in the preselect window, the system will already by tracking the preselected station so that there will be no delay when that frequency is transferred to active.

NAV tuning, which normally also selects the associated DME frequencies, is discussed under "Primus Remote Radio System-NAV" in this section. Special tuning procedures applicable to DME, which are in addition to the NAV tuning, will be discussed.

Two DI-850 indicators are installed; one on the pilot's and one on the copilot's instrument panel. DME information is presented on the DI-850 DME indicator and, when selected on the DC-550 display controller, on the pilot's and copilot's EHSIs. The channel (CH) button allows selection of NAV1 or NAV2 on either DI-850 indicator; each indicator can be selected to its own side or on the opposite side. Selections on the CH button on the indicator will not affect the selection(s) made on the DC-550 display controller, which controls the display on the respective EFIS. A selection on one DI-850 indicator will not effect the selection on the opposite indicator.

NAV1 or NAV2 will be annunciated on the top line of the indicator to indicate which NAV1s being displayed and computed. If the DME is being held, HLD is annunciated on the top line along with NAV1 or NAV2 to indicate which channel the DME is holding. When a station is being held, the regular functions are selectable on the DI-850 indicator and information will be computed from the station identified by "H" on the DME line of the RMU; however, after 15 seconds, the DI-850 annunciation will revert to identifier.

The select (SEL) button on the indicator is used to cycle the display on the right side of the readout through ground speed, time-to-station, and IDENT functions. If HOLD is selected on the DME, the function will return to IDENT in 15 seconds if any other function is selected.

The DME has a "split tuning" mode which operates somewhat like conventional HOLD functions, but provides other options. Pressing the DME button on the bottom of the RMU will

divide the NAV window into two windows. The top window will remain the active VOR frequency. H will be annunciated on the bottom line, indicating that the DME frequency is holding with the active frequency which is displayed on the top line. The bottom line will be labeled DME and will display the active frequency shown in VHF (VOR) format. The DME may then be tuned by pressing the line select key and changing it to a new channel. Pressing the DME button again will cause the DME (lower) window to change to a TACAN channel presentation. TACAN channels, along with their related W, X, Y, and Z channelization nomenclature will then be tunable with the tuning knobs. The DME function of all 126 TACAN channels may be tuned. No azimuth information is received in this mode. A third press of the DME button causes the NAV window to return to its normal active/preset presentation and the DME will resume tuning with the active frequency.

DME information is displayed on the pilot's and copilot's EHSIs by pressing the NAV button on the DC-550 display controller. Pressing the NAV button alternately selects NAV1 and NAV2 for display. If both NAV receivers are selected to the same NAV source, the NAV annunciations (VOR1, VOR2) on the EHS] will be in amber. The selected DME will always be the same as the NAV source (VOR). If no DME information is available, the DME readout will display amber dashes.

STANDBY RADIO CONTROL UNIT (SRC)

The STANDBY RADIO CONTROL UNIT (SRC) is normally located on the center instrument panel to the right of the engine gauges (Figure 16-14). It may be used in two modes: normal and emergency. The modes are selected by means of the mode switch on the SRC. The mode selections cycle as the switch is turned. In the emergency mode, EMRG is displayed vertically along the top right edge of the display. The SRC is powered from the the emergency DC bus through the NAV1 circuit breaker and receives power any time the battery switch is in the BATT or EMER position.

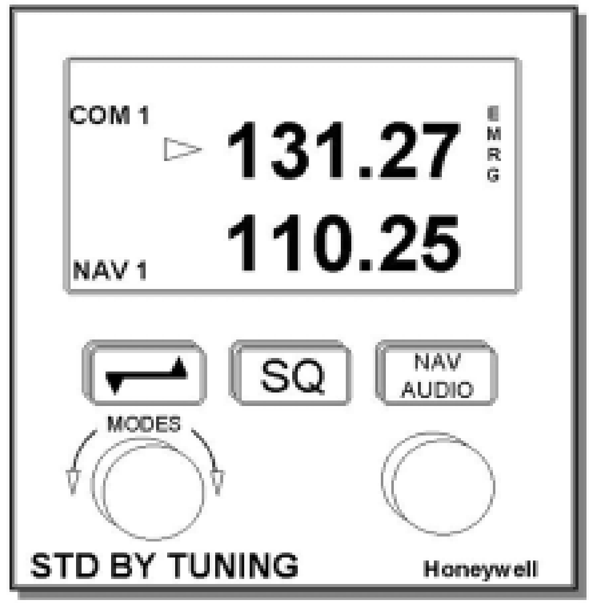


Figure 16-14 Standby Radio Control Unit (SRC)

In normal mode, the SRC acts as an additional tuning source for the radio system. COM1 and NAV1 may be tuned by the SRC in this mode. The SRC verifies that the COM1 RCZ850 or the NAV1 RNZ-850 (integrated COM and NAV units, respectively) are tuned to the correct frequency by checking the frequency echoed on the radio service bus (RSB). If the tuned frequency is incorrect, the frequency displayed on the SRC will be dashed out. If RMU1 is illuminated, the frequency change will appear in the active display. In normal mode, the radios which are tunable by the SRC (COM1 and NAV1) may also be tuned from the RMU1. If tuned from the RMU, the frequency will also be tuned on the SRC.

In emergency mode, operation of the SRC is identical on the part of the operator. The internal tuning of the system differs in that it does not read and compare frequencies on the RSB; whatever frequencies are set in the SRC are transmitted to the appropriate NAV or COM (RMU1) and that frequency is tuned.

When tuning the SRC, COM frequencies are displayed on the top line and NAV frequencies on the bottom. An arrow cursor, which ap-

pears in the left of the displayed frequencies may be toggled between the NAV and COM frequencies by pressing the double arrow (transfer) switch. The line on which the arrow appears is then tunable by the tuning knobs on the SRC.

The SQ pushbutton toggles the COM squelch open and closed. When the squelch is open, SQ is annunciated in the right center portion of the display.

When the EMER button is selected on the audio panel, the NAV AUDIO pushbutton toggles the NAV AUDIO on and off. When NAV AUDIO is on, it is summed in the COM audio. NAV AUDIO will be annunciated at the center left of the display.

Anytime the COM transmitter is being keyed, the TX annunciator in the center of the display will appear.

HONEYWELL PRIMUS II - AUDIO CONTROL UNIT

Two Honeywell Primus II digital audio control units are supplied with the Honeywell Primus II remote radio system. Digital transmission of audio from remote units to the audio panels differs from conventional audio systems in that it requires one twisted pair of wires rather than many twisted pairs to achieve the same performance.

The panels have three rows of combination audio ON-OFF switches and volume controls. The small round knobs serve as audio on-off switches when pressed (Figure 16-15). When the switch is latched in, the audio for the particular receiver it serves will be off. When pressed again, the switch will move outward turning the audio on. When the audio is on, the knob of the switch may be used as a volume control. Turning it clockwise will increase the volume; counterclockwise will decrease it.

Two larger knobs on the lower part of the control panel serve as volume controls for the pilot's/copilot's speaker and headset respectively. These knobs are in series with the smaller individual volume controls. This allows a volume selection to be made on the individual radio volume control, and then a final overall selection to be made by means of the speaker or headphone control, resulting in a more flexible individual control of all available audio signals.

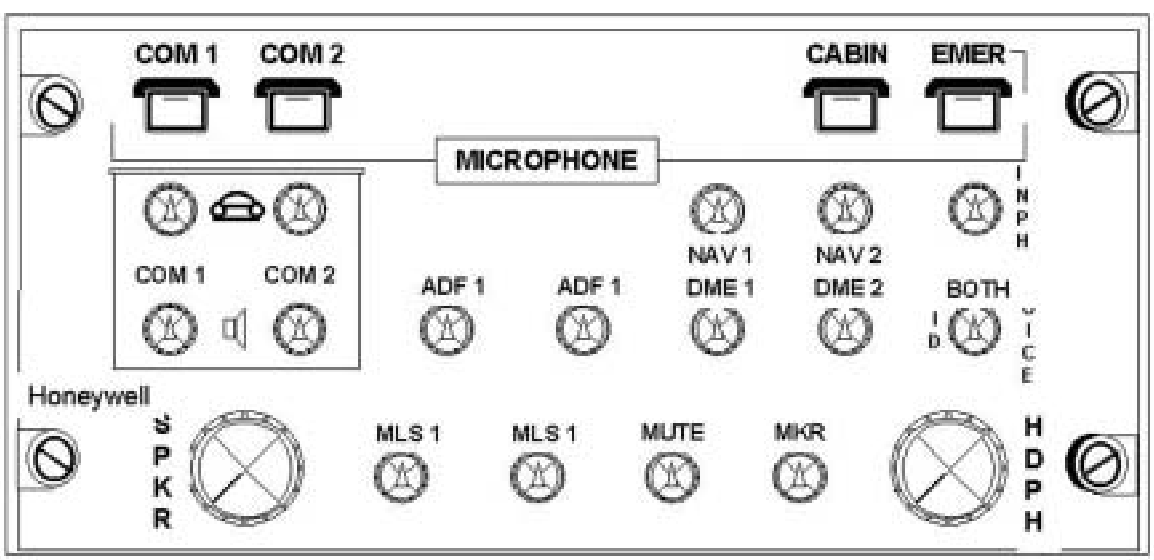


Figure 16-15 Audio Panel

A row of microphone selector buttons (push-push latching switches) is located across the top of the control panel. These buttons connect the pilot's or copilot's microphone to the selected transmitter. The receiver for the selected radio or interphone will also be selected regardless of the audio on-off switches selection. For night operation, a light above the microphone selector button is illuminated.

When depressed, the emergency COM (EMER) microphone switch, located at the upper right corner of the audio panel, connects COM1 transceiver directly to the aircraft microphone and headphone. All electronic circuitry is eliminated and all other audio panel modes are disabled in this mode. NAV1 audio will also be directed into the headset controlled by the panel on which EMER is activated, if NAV AUDIO is selected on the SRC.

RADIO ALTIMETER

The Collins ALT-55B radio altimeter displays radio altitude at all times up to an absolute altitude of 2,500 feet. The system becomes operational when the airplane electrical system is powered up, and it remains operational throughout the flight. Radio altitude is displayed in the bottom center of the attitude sphere in the EADI displays.

The altitude display in the EADIs operates from -20 to 2,500 feet. Between 200 and 2,500 feet, the display is in ten-foot increments. Below 200 feet, it is in 5-foot increments. Above 2,500 feet, the display disappears.

Decision height (DH) selection is displayed digitally in the lower right side of the EADI display. It is selected by means of the DH/TST knob on the DC-550 display controller. The EADI decision height range is from 0 to 990 feet in 10-foot increments. Full counterclockwise rotation of the DH/TST knob on the DC-550 display controller removes the DH display. A decision height warning horn sounds when the airplane reaches the decision height set on the pilot's EADI. The tone fades as the airplane descends through the altitude.

The decision height warning horn is controlled only by the DH setting in the pilot's EADI. The copilot's EADI decision height selection has no effect on the sounding of the DH warning horn. When the airplane descends below an altitude of 100 feet above selected decision height, a white box appears in the upper left side of the EADI. When the decision height is reached, an amber DH appears inside the box. The display flashes for ten seconds and then goes steady.

A "low altitude awareness display", which is a brown strip along the right side of the PFD, is used as a visual annunciation of the airplane's nearness to the ground. The low altitude awareness display is inside the bottom part of the altitude display and begins to appear when an altitude of less than 550 feet is reached. At touchdown, the low altitude awareness display reaches the horizon line.

If radio altimeter information is invalid, the radio altitude display will be amber dashes.

Functional testing of the radio altimeter system and the EADI display digital readout is accomplished on the ground by depressing the TEST button on the DC-550 display controller. The following displays occur: a radio altitude of 50 ±5 feet is indicated until the button is released, at which time the actual altitude is displayed. The decision height window displays dashes when the TEST button is held down and then displays the current set altitude for the remainder of the test. The radio altimeter TEST cannot be accomplished when the APP, CAP function of the flight director is in operation. The horn check depends on the DH altitude set on the pilot's EADI display.

LOCATOR BEACON

The ELT 110-4 Emergency Locator Transmitter (ELT) system is an emergency transmitter designed to assist in locating a downed airplane. The transmitter has a self-contained battery pack which must be changed every three years or after a cumulative total of one hour of operation. The system is activated

automatically by an impact of 5.0 +2/-0 Gs along the flight axis of the airplane, or manually by a remote ON/OFF switch located forward of the pilot's circuit breaker panel (Figure 16-16). When the transmitter is activated, a modulated omnidirectional signal is transmitted simultaneously on emergency frequencies 121.50 and 243.00 MHz. The modulated signal is a downward-swept tone signal starting at approximately 1,600 to 13,300 Hz and sweeping down every two to four seconds continuously and automatically.

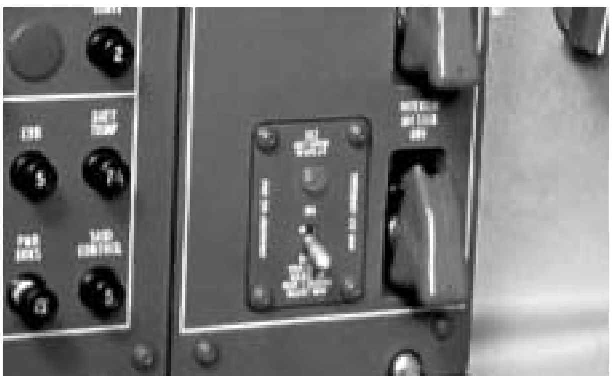


Figure 16-16 Emergency Locator Beacon (ELT)

The transmitter ON/OFF switch is normally left in OFF. ON position, the impact switch is bypassed and the emergency signal is transmitted.

The ELT incorporates a test feature that can be activated as follows:

1. Ensure the master avionics switch is ON.

2. Tune radio to 121.5 MHz

3. Place the ON/OFF switch ON for three sweeps of the receiver (approximately one second) and then to OFF. Ensure the amber indicator light illuminates immediately and begins flashing. If the test light does not illuminate immediately, the test failed.

PULSE EQUIPMENT

WEATHER RADAR-PRIMUS 880 COLORADAR

The Primus 880 ColoRadar system is an X-band alphanumeric digital radar with a display designed for weather location and analysis, and ground mapping (Figure 16-17). The system can be operated in conjunction with the EFIS and the MFD equipment to provide radar video displays. Storm intensity is displayed at five color levels, with black representing weak or no returns and green, yellow, red, and magenta showing progressively stronger returns. In the ground mapping mode, levels of returns are displayed as black, cyan, yellow, and magenta. The system consists of a receiver-transmitter antenna in the nose section and a controller. Some functions of the MFD system and the EFIS interface with the radar. Consult

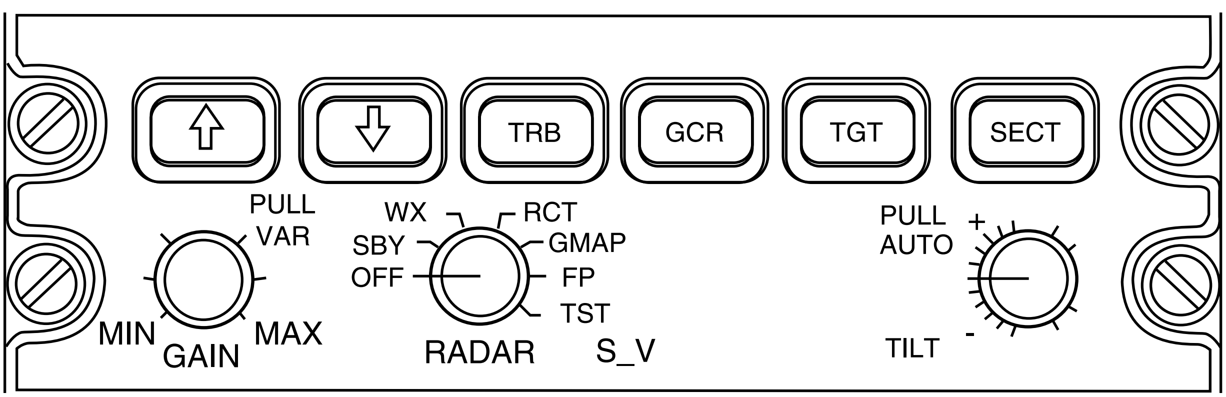


Figure 16-17 Primus 880 Coloradar Controller

the Airplane Operating Manual and vendor handbooks for operating instructions.

TRAFFIC ALERT AND COLLISION AVOIDANCE SYSTEMS (TCAS II) (OPTIONAL)

TCAS II detects and tracks aircraft in the vicinity of your own airplane. It interrogates the transponders of other aircraft and analyzes the signals to determine range and bearing, and relative altitude if it is being reported. It then issues visual and aural advisories so that the crew may perform appropriate vertical avoidance maneuvers.

The following information is generated and considered by the TCAS II in making a decision as to whether an aircraft which returns a signal constitutes a threat or not: range between your airplane and the intruder, relative bearing of the intruder, altitude and vertical speed of the intruder (if it is reporting altitude), and the closing rate between your aircraft and the intruder.

TCAS II is an independent airborne system. It is designed to act as a backup to the Air Traffic Control system and the "see-and-avoid" concept. TCAS consists of six airplane mounted antennas, a TCAS computer unit, and dual Mode S transponders; displays and controls are located in the cockpit. The following options are operational:

1. The TCAS is wired to display all traffic full time on the MFD display.

2. The TCAS display range is pilot selectable.

3. The TCAS system will automatically be in "TA ONLY" and not in standby when on the ground (and the TCAS system is active). Pilot selectable self-test in not inhibited in flight.

4. A Test Pattern is displayed on the MFD/PFD during pilot initiated TCAS II self-test.

TCAS has a surveillance volume defined by a minimum horizontal radius of 14 nautical miles and a minimum vertical range of +/- 12,700 feet. TCAS continually surveys the airspace around an airplane, seeking replies from other airplanes in the vicinity via their ATC transponders. The transponder replies are tracked by the TCAS system. Flight paths are predicted based upon these tracks. Flight paths predicted to penetrate a collision area surrounding the TCAS airplane are annunciated by TCAS.

TCAS generates two types of annunciations: a Traffic Advisory (TA) and a Resolution Advisory (RA). The airspace around the TCAS airplane can be divided into caution and warning areas. The physical dimensions of these areas are time-based (35-45 seconds TA, and 20-25 seconds RA) and vary as a function of horizontal and vertical closure speed and distance from an intruder airplane.

The TA display identifies the relative threat of each aircraft which could present a traffic conflict (intruder), by using various symbols and colors. TCAS II also provides several appropriate synthesized voice announcements which are used to alert traffic and to notify them of a recommended avoidance action. The TCAS II system is compatible with both current and planned ATC systems and operates independently of them. It has the capability to monitor two or more TCAS II-equipped aircraft by means of their mode S transponders and to coordinate their maneuvers.

The two types of cockpit display are the Resolution Advisory (RA) and Traffic Advisory (TA). The RA display is incorporated into the vertical speed indicator (VSI) display on the primary flight display (PFD). By illuminating red and green arcs around the display dial, it presents the required rate or limitation of climb or descent to avoid a possible collision. The resolution advisory (RA) is based on the expectation that the crew will

comply within 5 seconds. The system requires 2 1/2 seconds to show an increase or a reversal to an RA, In order for the system to generate an RA, the intruder must be reporting altitude; if an altitude is not being reported, the advisory will be limited to a TA.

The TA display on the MFD shows the intruding aircraft's relative position and altitude, with a trend arrow to indicate if it is climbing or descending at greater than 500 feet per minute. This display is provided at the bottom area of the MFD, which is reserved for TCAS presentation when TCAS is selected for display. AUTOMATIC "pop up" display or MANUAL display on the MFD is programmed through the RMUs.

Normally a TA precedes an RA by 15 seconds if an RA is going to ensue from the computation of closure rate, heading, rate-of-climb/descent, etc., of the intruder. Depending upon altitude, the system presents a traffic alert display, accompanied by an aural "Traffic Traffic," when the time to the closest point of approach is between 35 and 45 seconds. The crew should attempt to gain visual contact with the intruder and be prepared to maneuver. The crew should take no evasive action based solely on the TCAS II traffic display.

TCAS II can track as many as 45 aircraft at one time and display up to 30 of them. It can coordinate a resolution advisory for as many as three intruders at one time. The advisories are always generated considering the least required amount of deviation from the flight while providing a safe vertical separation.

TCAS II does not replace ATC procedures and the existing "see-and-avoid" concept; however, if ATC communications are temporarily lost, TCAS II adds a significant backup capability for collision avoidance, and can also enhance safety of flight in crowded terminal areas, under both VFR and IFR conditions.

TCAS continuously calculates tracked airplane projected positions. TAs and RA's are therefore constantly updated and provide real-time advisory and position information.

Once the flight path of the intruder no longer conflicts with the collision area of the TCAS airplane, TCAS announces "Clear of conflict." The flight crew should then return to the original clearance profile.

TCAS generates TAs and RA's against intruder airplanes with ATC transponders replying in Mode C and Mode S. TCAS requires altitude information from intruder airplanes to generate RA's. TCAS can provide only TAs for intruder airplanes whose transponders reply in Mode A (non altitude reporting).

CAUTION

TCAS cannot provide an alert for traffic conflicts with airplanes without operating transponders.

If an installation includes a windshear warning system and/or a ground proximity warning system, in conjunction with the TCAS II system, the aural warning priority is as follows:

1. Windshear warning
2. Ground proximity warning
3. TCAS II warning

ALLIED SIGNAL ENHANCED GROUND PROXIMITY WARNING SYSTEM (EGPWS) (OPTIONAL)

The Allied Signal Enhanced Ground Proximity Warning System (EGPWS) provides visual and aural warnings of terrain in the following Basic GPWS modes:

1. Excessive rate-of-descent with respect to the terrain (Mode 1).
2. Excessive closure rates to terrain (Mode 2).

3. Negative climb before acquiring a predetermined terrain clearance after takeoff or a missed approach (Mode 3).

4. Insufficient terrain clearance based on flap configuration (Mode 4)

5. Inadvertent descent below glide slope (Mode 5)

6. Minimums callout upon reading DH (Mode 6).

7. SMART 500 callout - Altitude callout at 500 AGL (Mode 6).

8. Excessive bank angle alerting (Mode 6).

9. Windshear Warning and Windshear Caution Alerts (Mode 7)

In addition, the Enhanced Ground Proximity Warning System provides the following terrain map enhance modes:

1. Terrain Clearance Floor Exceedance.

2. "Look-Ahead" Cautionary Terrain Alerting and Warning Awareness.

3. Terrain Awareness Display. EGPWS provides display of approximate terrain and obstacles. The terrain display is color- and intensity-coded (by density) to provide visual indication of the relative vertical distance between the airplane and the terrain.

Aircraft equipped with the optional EGPWS have a red PULL UP, amber BELOW G/S and G/S CANCELED, and green TERR INHIB annunciators located directly above the Radio Management Units.

An aural "Pull up" warning sounds if any of the terrain proximity mode windows is entered as noted above. During ILS glideslope approaches, the below-glide-slope warning may be canceled if desired (runway in sight and deliberately flying below glide slope for landing) by depressing the BELOW G/S switchlight and illuminating the lower half labeled "G/S CANCELED."

The switchlight labeled "GPWS FLAP OVRD/ACTIVE" is provided to disable the flap configuration input to prevent nuisance warnings when landing with less than full flaps (aural and visual warnings would normally be initiated at 200 feet with less than full flaps). The lower GPWS TEST/GPWS INOP lights are provided to perform functional tests and provide indication of system malfunctions. The visual and aural warnings are initiated as the rotary test switch is positioned to ANNU.

Self-Test

The system is tested by pushing the GPWS TEST button and holding it in for the duration of the test (located above the RMUs on the center instrument panel). The following aural messages will be heard and annunciators displayed during the test:

1. GPWS FAIL and WSHR FAIL (AMBER message in MFD).

2. Aural "GLIDESLOPE" is heard and boxed GND PROX appears in PFD ADI.

3. Aural "PULL UP" is heard and boxed red PULL UP appears in PFD ADI.

4. Aural "WINDSHEAR-WINDSHEAR-WINDSHEAR" is heard and boxed red WINDS HEAR followed by boxed amber WINDSHEAR appears in PFD ADI (Mode 7 only).

5. Aural "TERRAIN-TERRAIN, PULL UP-PULL UP" is heard and boxed red PULL UP appears in PFD ADI.

NOTE

GPWS self-test is inhibited in flight.

UNIVERSAL AVIONICS TERRAIN AWARENESS WARNING SYSTEM (TAWS) (OPTIONAL)

The TAWS system configured with the UNS-1Csp FMS provides terrain situational awareness relative to current and predicted aircraft position as well as advanced ground proximity warning. The system provides alert information both visually and aurally.

TAWS provides displays of terrain with flight path intent information in several views, Map, Profile and 3-D views.

The terrain data base is stored in flash memory and contains a data point approximately every one-half mile worldwide, one at least one-forth mile between 30 degrees S and 40 degrees N latitude where most aircraft operate, and at least one-tenth mile at mountainous airports.

TAWS provide alerts in accordance with standard GPWS functionality modes.

More complete information is contained in the Citation Excel Airplane Flight Manual (AFM), Supplement 31 and the Universal Avionics TAWS Operator's Manual

AREA NAVIGATION

UNIVERSAL UNS-1C(SP) FLIGHT MANAGEMENT SYSTEM (FMS)

Navigation Management

Universal Avionics Systems UNS-1Csp Flight Management System is a centralized control and master computer system, designed to consolidate and optimize the acquisition, processing, interpretation and display of certain aircraft navigational and performance data. The UNS-1Csp FMS system may be installed as GPS only or multi-sensor system. Digital Air Data information (including baro-corrected altitude and true airspeed) and heading input is required in all installations.

The Navigation Computer Unit (NCU) has multiple ports through which data from external sensors can be received. The long range navigation sensors that may be accommodated include Inertial, GPS, and Loran C. A radar joy stick for remote way point entry can also be accommodated. When a DME interface is included, the DME input is considered a short range sensor and is from a multi-channel scanning DME. All DME stations within approximately 250 nm of the aircraft position are scanned, and up to 15 are continuously tracked. If a VOR input is provided, it will be used for VOR/VOR-DME /RNAV approaches when GPS integrity does not meet integrity requirements. VOR will be used enroute as a last resort sensor.

Each individual navigational sensor is specifically designed for primary navigation. The FMS system takes advantage of a particular sensor's good properties while minimizing its liabilities. The system processes multiple range information from the DME, True Air Speed data from the Air Data Computer, velocity and position information from the long range navigation sensors, and aircraft heading, in order to derive one Best Computed Position (BCP). This is accomplished by a Kalman Filtering of the various sensors.

Navigation Data Base

The FMS contains a memory capacity up to 100,000 waypoints in a non-volatile flash RAM (no battery required). The stored JEPPESEN data base provides the capacity for complete coverage for SID's, STARS, Approaches, High/Low Airways, Navaids, IFR intersections, and airports with runways longer than 4,000 feet with IFR Approach in the worldwide data base and 2,000 feet in the regional data base. World-Wide or Regionalized data base subscription services are also available.

Dual Cycle Data Base

The UNS-1Csp Contains a dual memory bank which is capable of holding the next cycle data base before it becomes effective. When the current data base expires, the next cycle data base will automatically become active. When the FMS is powered up, one of the functions of the self test is to determine which data base is active by comparing effective dates and expiration dates.

Pilot Data Storage

Extensive memory space is also allocated for Pilot Defined Data. The system can store in memory up to 200 pilot created flight plan routes comprised of up to 98 waypoints each, and up to 200 pilot defined waypoints, 100 arrivals/departures, 100 approaches, 100 runways, 100 airports, 100 alignment points and 25 radar waypoints. Once stored, pilot defined data is easily accessed and may be added to the flight plan with a few simple key strokes.

Company Routes Data

A protected company routes data base can be created using Universal's Offline Flight Planning Program and may be contained on up to four disks. A total of up to 2000 routes and 250,000 route elements may be stored. Each route will consist of at least one but not more than 98 legs (route elements). Route elements reference waypoints, airways and terminal area procedures (SIDS, STAR's and approaches) from the Jeppesen navigation data base by use of reference pointers. Company routes can be given names of up to eight characters in length. A company routes data base will allow routes to be viewed and copied to the active flight plan, but the routes themselves cannot be modified by the flight crew. A configuration module option has been added which will cause all pilot data to be cleared at each power on cycle. This feature is mainly for airline operations in which no leftover pilot data are allowed in the FMS between power cycles.

A company route data base must be loaded with the same Jeppesen data which was used for its creation. Company routes data base disk(s) are loaded through the Data Transfer Unit (DTU) by using the same procedures as loading Jeppesen data disks.

Off Line Flight Planning

The Data Transfer Unit (DTU) is capable of both reading and writing data. Your FMS created flight plans can be downloaded to diskette for review in the comfort of your office or flight department using a compatible desktop computer. Modify or add to your pilot defined data and upload it via diskette to the FMS. Determine ETEs while varying routes. Study the effect of winds. Compute fuel requirements and reserves. You can even upload flight plans from several leading flight planning services.

Fuel Management Monitoring

The fuel management function enables the pilot to plan fuel requirements while on the ground. Input from the aircraft's fuel flow sensors, along with pilot supplied data, enable the FMS to calculate and display significant real-time fuel management information throughout the flight. While enroute, alternate destinations, ground speed or fuel flow values may be temporarily entered to check various "what if' scenarios.

Frequency Management

With compatible radios and an optional Radio Tune Unit (RTU), the communications transceivers, VOR receivers and TACAN (if installed) may be tuned through the CDU (Figure 16-18). This function provides 4 presets for each radio which may be stored by alpha identifiers or tuning from a list of identifiers and instant recall of the last frequency entered.

Laterial Guidance and Steering

Using the best computed position and the flight plan described by the pilot, the FMS computes Great Circle route legs to navigate the

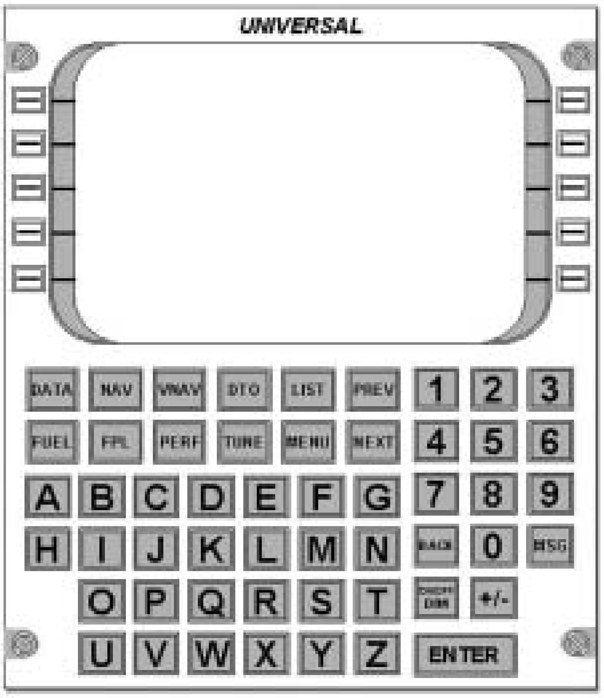


Figure 16-18 Universal UNS-1Csp (CDU)

aircraft along the programmed flight path. Optional maneuvering procedures include; present position direct to (DTO), Pseudo - VOR, FMS Heading, Selected Cross-Track (STX), Holding Patterns, SIDS, STARS, and Approaches, are provided. The FMS provides desired track, bearing, cross-track, lateral deviation, and related data to the flight guidance system for the EFIS displays, and roll steering commands for the autopilot/flight director system. The system anticipates leg changes and provides "smart turns" to eliminate "S" turning. . Pilots can designate way points for over flight. The roll steering command is gain-scheduled based on altitude for gentle turns. The bank limit is configured at time of installation to match FGS limits.

Vertical Guidance

A 9-waypoint enroute vertical navigation descent profile may be programmed. The FMS will display the vertical speeds required based on present ground speed to obtain the target altitudes at the VNAV waypoints. A computed Top of Descent Point, based on target vertical speed, is displayed. When the Top of Descent Point is reached, the system provides vertical deviation from the vertical flight path for output to a vertical deviation pointer. The flight path angle (FPA) is limited to settings in the configuration module set at time of installation. FMS VNAV may be coupled to the Flight Director and Autopilot for automatic descent flight guidance.

In the approach mode, a pseudo glide slope may be defined. Deviation signals similar to an ILS glide slope are available for output to the EFIS and flight guidance systems.

Complete information regarding programming and operating the UNS-1Csp is contained in the following manuals:

- *Citation Excel AFM, Supplement 1*
- *Universal Avionics Operator's UNS-Csp Training Manual*
- *Universal Avionic's Operator's UNS-1Csp Manual*

COCKPIT VOICE RECORDER (CVR)

The A-200S cockpit voice recorder (CVR) provides continuous recording of the last 30 minutes of all voice communications and aural warnings originating in the cockpit, as well as sounds from various warning horns. The system requires main DC power and is protected by a 5-amp circuit breaker on the LH CB panel (CVR).

A sensitive microphone is located to the left of the engine fire tray. The recorder is energized anytime the battery switch is in the BATT position. A control panel located on the copilot's lower instrument panel (Figure 16-19), contains a TEST button and an ERASE button.

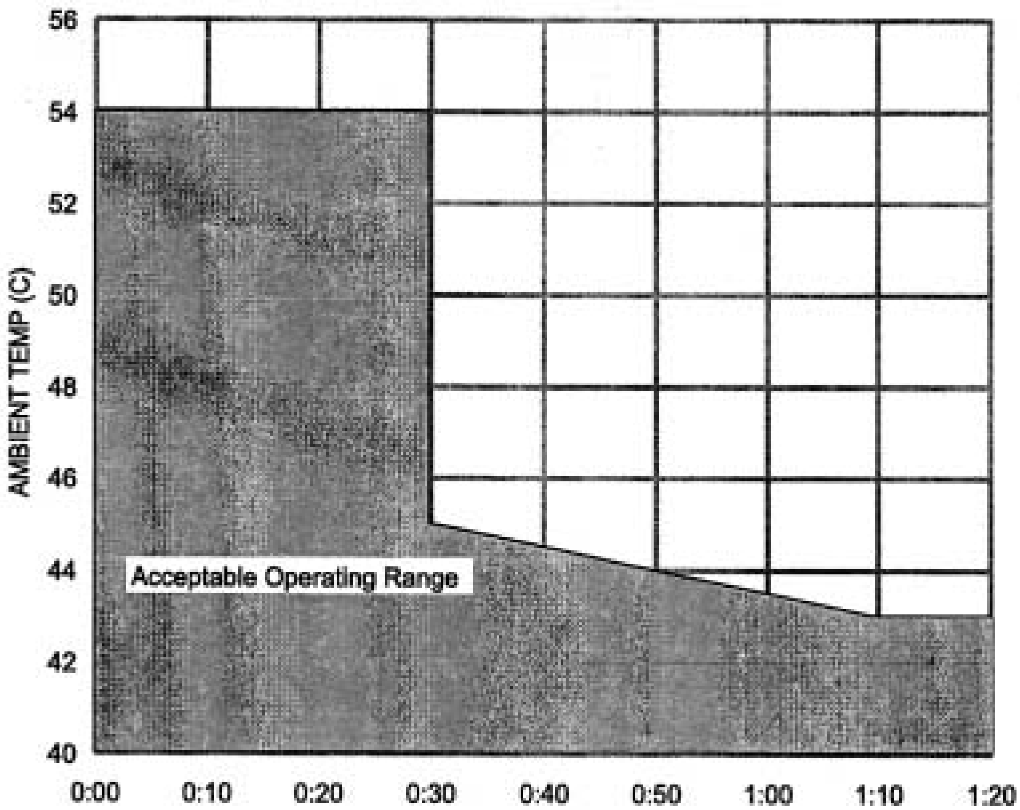


Figure 16-19 Avionics/Electrical Operating Time (Hrs:Mins)

Holding the TEST button down for 5 seconds will cause a green light on the panel to illuminate indicating the CVR is functional.

To erase the CVR the airplane must be on the ground and the cabin door open. Pressing the ERASE button for approximately 2 seconds will cause the entire record to be erased.

STATIC DISCHARGE WICKS

A static electrical charge, commonly referred to as "P" (precipitation) static, builds up on the surface of an airplane while in flight and causes interference in radio and avionics equipment operation. The static wicks are installed on all wing and empennage trailing edges, and dissipate static electricity in flight.

NOTE

Do not wax the aircraft with products containing silicones. They can contribute to P-static buildup, especially if the surfaces are buffed to produce a shine.

There are a total of 20 static wicks:

- One on each wingtip
- Four on each wing trailing edge outboard of the aileron
- One on the trailing edge of each aileron
- Two on the trailing edge of each elevator
- Two on the upper trailing edge of the rudder
- One on the top of the rudder
- One on the tail stinger

CAUTION

IF ANY STATIC WICK IS MISSING FROM THE ELEVATOR, RUDDER, OR AILERON, IT SHOULD BE REPLACED BEFORE FLIGHT TO ENSURE PROPER CONTROL SURFACE BALANCE. ONE WICK ONLY ON EACH WING AND THE EMPENNAGE MAY BE MISSING OR BROKEN FOR DISPATCH (17 MINIMUM), BUT THERE IS A RISK OF PRECIPITATION STATIC.

ONE STATIC WICK ONLY MAY BE MISSING OR BROKEN FROM EACH OF THE FOLLOWING AREAS: LEFT AND/OR RIGHT WING (WING TRAILING EDGE OR WINGTIP OR AILERON); EMPENNAGE (VERTICAL STABILIZER OR RUDDER).

LIMITATIONS

Avionics ambient temperature limits:

AUTOPILOT

1. One pilot must remain in their seat, with seatbelt fastened, during all autopilot operations.
2. Autopilot operation is prohibited if any comparison monitor annunciator illuminates in flight.
3. Minimum autopilot engagement height is: 1,000 feet AGL–Enroute; 300 feet AGL–Nonprecision Approach; 180 feet AGL–Category I ILS Approach.

HONEYWELL PRIMUS-1000 FLIGHT GUIDANCE SYSTEM

1. The Pilot’s Manual for the Honeywell P-1000 Integrated Avionics System for the Cessna Citation Excel airplanes equipped with IC-600, part number A28-1146-120-00, revision 0, or later applicable revision, must be immediately to the flight crew. For airplanes equipped with IC-615, part number A28-1146-137-00, revision 0, or later applicable revision, must be immediately to the flight crew
2. Category II approaches are not approved.
3. EFIS ground operation with the **RADOME FAN FAIL** annunciator illuminated is limited to 30 minutes or until either **IC-1** or **IC-2 HOT** annunciator illuminates, whichever occurs first.
4. Dispatch is prohibited if either the **IC-1** or **IC-2 HOT,** annunciator is illuminated.
5. Dispatch in instrument meteorological conditions is prohibited with the **RADOME FAN FAIL** annunciator light illuminated. Dispatch in visual meteorological conditions is allowed with the **RADOME FAN FAIL** annunciator illuminated, provided the DISPLAY GUIDANCE COMPUTER COOLING FAN FAILURE abnormal procedures are followed.
6. Dispatch is prohibited following a flight where either a **IC-1 HOT** or **IC-2 HOT** annunciator light was illuminated, until the condition is identified and corrected.
7. The pilot’s and copilot’s PFDs must be installed and operational in the normal (non-reversionary) mode for takeoff.

8. The P-1000 system must be verified to be operational by a satisfactory preflight test as contained in Section III of the AFM Normal Procedures.

9. Reversion of both PFDs to the MFD is prohibited.

10. VOR approaches without a valid DME signal are prohibited with autopilot coupled or with flight director only.

NOTE

Enroute VOR navigation without a valid co-located DME signal may result in significantly degraded course tracking when utilizing the flight director or autopilot. The flight crew should monitor the CDI for excessive deviation and select HDG mode as required to manually track the desired course.

STANDBY FLIGHT DISPLAY

1. A satisfactory preflight test must be accomplished on the STBY PWR system in accordance with Section III of the AFM Normal Procedures.

2. The standby flight display (including ATT,ALT and ASI) and HSI must be functioning prior to takeoff.

NOTE

To prevent precession of attitude information, once AHRS is selected in flight to DG MODE, AHRS must remain in DG MODE for the remainder of the flight. AHRS may be returned to SLAVE MODE after landing or prior to next flight if appropriate.

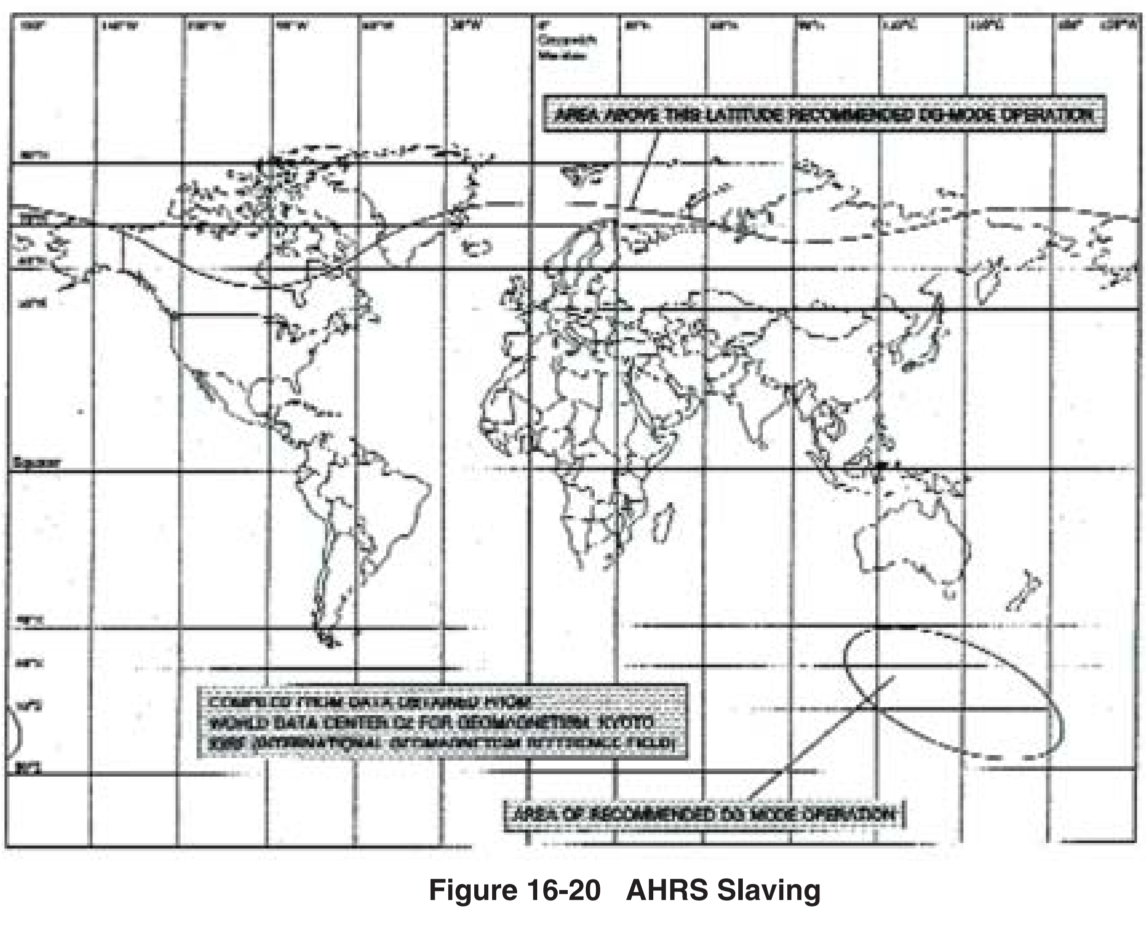


Figure 16-20 AHRS Slaving

CHAPTER 17
OXYGEN SYSTEMS

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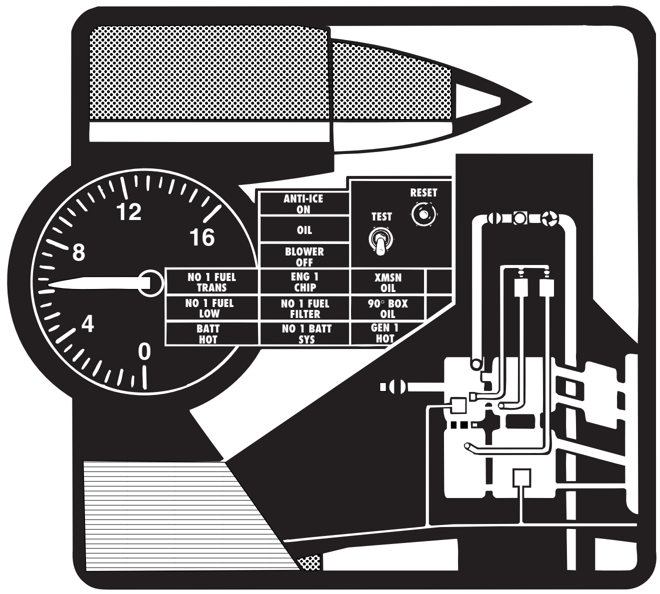
CHAPTER 17
OXYGEN SYSTEMS



INTRODUCTION

The Citation Excel is equipped with a supplemental oxygen system that can be utilized during emergency situations such as loss of cabin pressure, medical situations, or smoke concentrations. The flight station is equipped with quick-donning O_2 masks. The cabin has masks installed in overhead compartments for passenger use that may be deployed automatically if cabin altitude becomes excessive or may be dropped manually by the flight crew, if required.

GENERAL

The oxygen system consists of crew and passenger distribution systems. Oxygen is available to the crew at all times and is available to the passengers either automatically above a predetermined cabin altitude, or manually at any altitude by a cockpit control selector. The system is intended to provide an emergency oxygen supply if a loss of cabin pressure occurs, smoke is in the cabin or cockpit, and medical emergencies.

The system consists of an oxygen storage cylinder, pressure regulator, servicing fitting, crew and passenger masks, an altitude pressure switch, overboard discharge disc, bottle pressure gage, and a control selector on the pilot's console below the instrument panel.

COMPONENT DESCRIPTIONS

OXYGEN CYLINDER ASSEMBLY

The standard oxygen cylinder installed in the right side of the lower nose compartment (Figure 17-1) has a 49-cubic-foot capacity with an option for a 76-cubic-foot cylinder. A shutoff valve and pressure regulator, located on the cylinder, control the flow of oxygen to the distribution system. The shutoff valve is normally open and the regulator reduces line pressure to approximately 70 psi. The cylinder is serviced through the filler port in the lower aft sill of the right nose compartment door with aviators breathing oxygen.

PRESSURE GAGE

A direct-reading mechanical oxygen pressure gage is located on the right side of the copilot's instrument panel (Figure 17-2). The gage reads cylinder pressure any time the system is charged, regardless of the position of the shutoff valve on the cylinder. A fully serviced system should read 1,600-1,800 psi.

The system should be serviced any time gage pressure indicates out of the green arc. It must be serviced if pressure drops below 400 psi, and the system must be purged if the bottle is allowed to deplete to empty.

Figure 17-1 Oxygen Cylinder

Figure 17-2 O_2 Pressure Gage

CONTROLS

The PASS OXY selector on the pilot's console (Figure 17-3) controls oxygen flow to the passengers or restricts it to crew use only. OFF, AUTO, and ON positions allow automatic or mechanical actuation of the passenger distribution system as desired.

OVERBOARD DISCHARGE INDICATOR

A green overboard discharge indicator (disc) is located below the aft edge of the right nose compartment door (Figure 17-4). The disc provides a visual indication that an overpressure condition occurred in the cylinder and the bottle is empty. Maintenance must be performed before flight if the disc is missing.

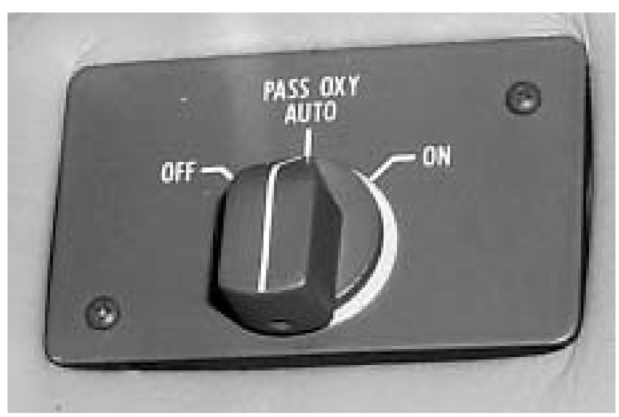


Figure 17-3 Oxygen Selector Switch

Figure 17-4 Overboard Discharge Indicator

OXYGEN MASKS

Crew masks stowed in a retainer just below each crewmember's side window (Figure 17-5) are quick-donning EROS masks with integral microphones and a regulator with three positions. After the masks are donned by the crew, it will be necessary to place the respective microphone switch located on the lower outboard side of each instrument panel from MIC HEAD SET to MIC OXY MASK to communicate.

Selecting EMER (on the masks) makes pressure breathing possible by providing a steady flow to the mask. EMER should be used if smoke or fumes are present.

In the 100% position, red switch (Figure 17-5), the user is assured oxygen is being received when there is no apparent restriction to breathing. 100% should be used during high cabin altitudes above 20,000 feet.

The NORM position is for diluter demand (oxygen mixing with ambient air). To conserve oxygen when using the crew masks, the regulators should be set to NORM if cabin altitude is below 20,000 feet.

The masks must be stowed properly in the receptacles to qualify as quick-donning masks. When using the mask with fumes or smoke present, select EMER to prevent smoke or fumes from entering a possible loose-fitting mask.

Passenger masks are stowed in overhead containers and can be dropped automatically or manually. Oxygen does not flow to the masks until a lanyard cord attached to the mask from a pin installed in the overhead compartment is pulled. The pin is inserted in the oxygen supply line to prevent oxygen from flowing to a mask that is not required in order to conserve oxygen.

Figure 17-5 Crew Oxygen Mask, Stowed

NOTE

All passenger masks drop during drop actuation, manually or automatically. Ensure passengers are briefed to pull their respective mask lanyard cords to receive oxygen.

If cabin altitude exceeds approximately 14,500 feet, the passengers masks will automatically drop (control switch, AUTO).

SYSTEM OPERATION

With the OXYGEN selector in the AUTO position, low-pressure oxygen at 70 psi is available to both crewmember's through outlets on the side consoles and to the solenoid valve on the oxygen selector (Figure 17-6).

The solenoid valve is normally spring-loaded closed, blocking flow to the passenger distribution system. If cabin altitude exceeds 14,500 feet, an altitude pressure switch energizes the solenoid valve open (Figure 17-6). Oxygen flowing into the passenger distribution system releases latches on the mask compartment doors, allowing the doors to open and the masks to fall out. If cabin pressure is restored to normal values, the solenoid valve is deenergized at 6,800 feet cabin altitude, shutting off oxygen flow to the passengers.

If a loss of DC power occurs, the solenoid valve wll remain closed regardless of cabin altitude. Placing the OXYGEN selector ON, with or without DC power, allows oxygen to flow through the manual control valve to the passenger distribution lines, unlocking the doors and causing all the passenger masks to drop.

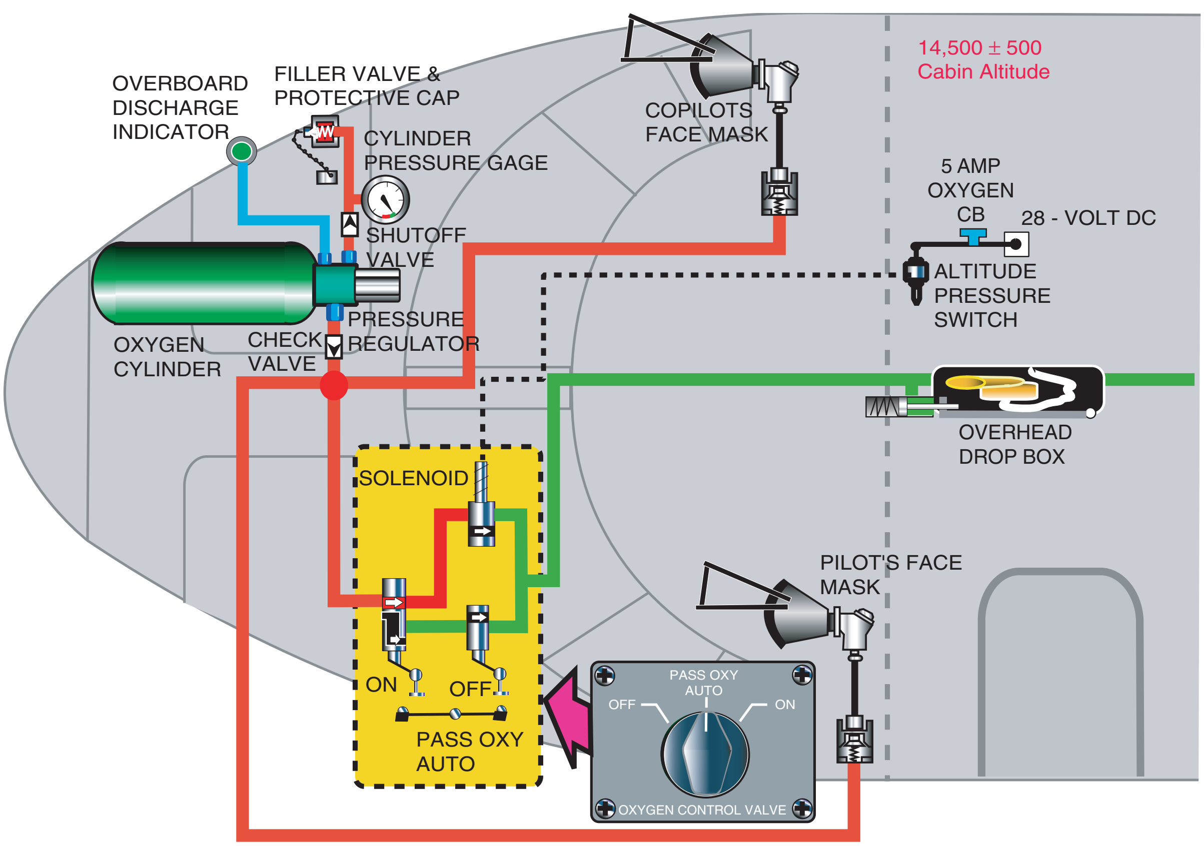


Figure 17-6 Oxygen System Schematic

The OFF (crew only) position manually blocks flow at the oxygen control valve, shutting off all flow to the passengers.

WARNING

NO SMOKING IS PERMITTED WHEN USING OXYGEN; OIL, GREASE, SOAP, LIPSTICK, LIP BALM AND OTHER FATTY MATERIALS CONSTITUTE A SERIOUS FIRE HAZARD WHEN IN CONTACT WITH OXYGEN.

WARNING

NO SMOKING WHEN OXYGEN IS BEING USED OR FOLLOWING USE OF PASSENGER OXYGEN UNTIL LANYARDS HAVE BEEN REINSTALLED.

Table 17-1 depicts the average time of useful consciousness (time from onset of hypoxia until loss of effective performance) at various cabin altitudes.

Table 17-2 depicts oxygen duration times.

Due to human physiological limitations, the passenger oxygen system is not certified for continuous operation above 25,000 feet cabin altitude and the crew oxygen system is not certified for continuous operation above 40,000 feet cabin altitude. Individual physiological limitations may vary. If crew or passengers experience hypoxic symptoms, descend to a lower cabin altitude.

NOTE

Cockpit masks are assumed to be at the normal setting at 20,000 feet cabin altitude with a respiratory rate of 10 liters per minute-body temperature pressure saturated and at 100% setting at and above 25,000 feet.

Table 17-1 AVERAGE TIME OF USEFUL CONSCIOUSNESS

15,000 to 18,000 feet	30 minutes or more
22,000 feet	5 to 10 minutes
25,000 feet	3 to 5 minutes
28,000 feet	2 1/2 to 3 minutes
30,000 feet	1 to 2 minutes
35,000 feet	30 to 60 seconds
40,000 feet	15 to 20 seconds
45,000 feet	9 to 15 seconds

LIMITATIONS

SUPPLEMENTAL OXYGEN SYSTEM

The following aircraft certification requirements are in addition to the requirements of applicable operating rules. The most restrictive requirements (certification or operating) must be observed:

Crew and passenger oxygen masks are not approved for use above 40,000 feet cabin altitude. Prolonged operation of passengers masks above 25,000 feet cabin altitudes is not recommended.

WARNING

PASSENGER MASKS ARE INTENDED FOR USE DURING AN EMERGENCY DESCENT TO AN ALTITUDE NOT REQUIRING SUPPLEMENTAL OXYGEN.

The pressure demand crew oxygen masks must be properly stowed in their containers to qualify as a quick donning oxygen mask.

NOTE

Headsets, eyeglasses, or hats worn by the crew will interfere with the quick-donning capabilities of the oxygen masks.

Table 17-2 OXYGEN SUPPLY

EROS CREW MASK AND 49-CUBIC FOOT CYLINDER

	AVAILABLE TIME IN MINUTES							
CABIN ALTITUDE	1 COCKPIT	2 COCKPIT	2 COCKPIT 2 CABIN	2 COCKPIT 4 CABIN	2 COCKPIT 6 CABIN	2 COCKPIT 8 CABIN	2 COCKPIT 10 CABIN	2 COCKPIT 11 CABIN
8,000	839	420	106	61	42	33	27	24
10,000	964	482	110	62	43	33	27	25
15,000	964	482	112	63	44	34	28	25
20,000	757	379	107	62	44	34	28	25
25,000	405	202	87	55	41	32	27	24
27,000	475	237						
29,000	523	261						
31,000	588	294						
33,000	663	332						
35,000	748	374						
37,000	851	426						
39,000	1037	518						

EROS CREW MASK AND 76-CUBIC FOOT CYLINDER

	AVAILABLE TIME IN MINUTES							
CABIN ALTITUDE	1 COCKPIT	2 COCKPIT	2 COCKPIT 2 CABIN	2 COCKPIT 4 CABIN	2 COCKPIT 6 CABIN	2 COCKPIT 8 CABIN	2 COCKPIT 10 CABIN	2 COCKPIT 11 CABIN
8,000	1308	654	165	94	66	51	41	38
10,000	1502	751	172	97	68	52	42	38
15,000	1502	751	175	99	69	53	43	39
20,000	1180	590	167	97	69	53	43	39
25,000	630	315	136	86	63	50	41	38
27,000	740	370						
29,000	815	407						
31,000	916	458						
33,000	1034	517						
35,000	1165	583						
37,000	1326	663						
39,000	1616	808						

QUESTIONS

1. The cockpit oxygen pressure gage reads:
   A. Oxygen pressure at the crew masks.
   B. Bottle pressure, electrically.
   C. Bottle pressure, mechanically.
   D. Requires DC power

2. Passenger masks are dropped as follows:
   A. Automatically with the PASS OXY selector in AUTO and cabin altitude exceeds 14,500 feet.
   B. If cabin altitude exceeds 13,500 feet, regardless of PASS OXY selector.
   C. PASS OXY selector ON regardless of altitude.
   D. A and C.

3. If DC power fails, placing the PASS OXY selector in:
   A. ON deploys the passenger masks, regardless of DC power on or off.
   B. ON deploys the passenger masks only if 14,500 feet cabin altitude is exceeded.
   C. OFF does not restrict oxygen to the crew; only if the cabin altitude is above 14,500 feet.
   D. None of the above.

4. The purpose of the altitude pressure switch is to:
   A. Bypass oxygen flows directly to the passengers regardless of the PASS OXY selector position.
   B. Open a solenoid at 14,500 cabin altitude, allowing oxygen flow to the passenger oxygen distribution system.
   C. Close a solenoid valve at 14,500 feet cabin altitude, stopping oxygen flow to the passengers.
   D Automatically if the PASS OXY selector is AUTO and the cabin exceeds 10,000 feet.

5. If normal DC power is lost with the PASS OXY selector in AUTO:
   A. The passenger masks will deploy immediately, regardless of the cabin altitude.
   B. The passenger masks cannot be dropped manually.
   C. The oxygen pressure gage on the copilot's panel will be inoperative.
   D. Automatic dropping of the passenger masks will not occur.

ANSWERS TO QUESTIONS

Chapter 2
1. C
2. B
3. A
4. C
5. D
6. C
7. B
8. D
9. B
10. D

Chapter 3
1. D
2. D
3. D
4. D
5. A

Chapter 5
1. D
2. C
3. B
4. A
5. C
6. A
7. D

Chapter 7
1. B
2. A
3. B
4. D
5. A
6. C
7. D
8. B
9. D
10. A

Chapter 8
1. C
2. D
3. A
4. D
5. B
6. D

Chapter 9
1. C
2. B
3. B
4. A

Chapter 10
1. A
2. B
3. C
4. D
5. D
6. B
7. C

Chapter 11
1. B
2. B
3. C
4. D
5. D
6. A

Chapter 12
1. B
2. D
3. C
4. D
5. A
6. C
7. B
8. D
9. A
10. A

Chapter 13
1. D
2. D
3. B
4. C
5. C
6. D
7. A
8. B
9. D
10. D
11. C
12. A
13. B

Chapter 14
1. B
2. D
3. A
4. B
5. C
6. A
7. B
8. C
9. C
10. C
11. B
12. B

Chapter 15
1. B
2. C
3. A
4. C
5. B
6. A
7. D
8. B
9. A
10. B
11. B
12. A
13. B
14. B
15. D

Chapter 17
1. C
2. D
3. A
4. B
5. D